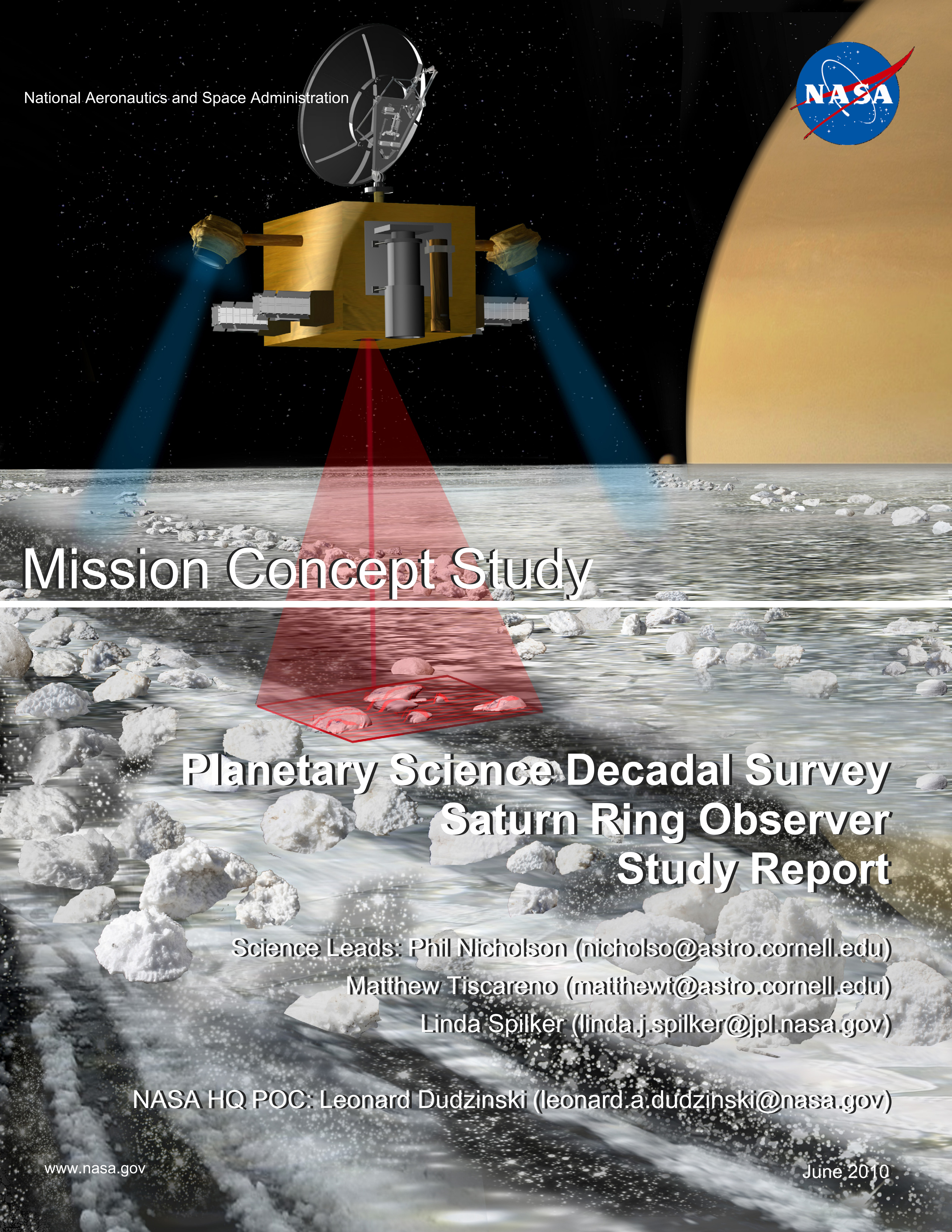

National Aeronautics and Space Administration

Mission Concept Study

Planetary Science Decadal Survey
Saturn Ring Observer
Study Report

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Data Release, Distribution, and Data Use Statements

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Planetary Science Decadal Survey

Mission Concept Study Final Report

Figures

Tables

Appendices

A. Acronyms

B. References and Bibliography

C. Ring Hazards

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Executive Summary

The Saturn Ring Observer (SRO) study was requested by the Giant Planets Panel of the 2012 Planetary Science Decadal Survey (PSDS). The panel specified two study objectives:

1. Investigate the method(s) by which a spacecraft might be placed in a tight circular orbit around Saturn, using chemical or nuclear-electric propulsion or aerocapture in Saturn's atmosphere. The critical issue is trajectory.
2. Identify technological developments for the next decade that would enable such a mission in the post-2023 timeframe (after the next saturnian equinox), with a particular focus on power and propulsion.

The "tight circular orbit" refers to a non-Keplerian orbit, displaced 2–3 km from the mean ring plane in the direction perpendicular to that plane. Since a spacecraft in such an orbit would appear to "hover" over the ring particles directly "beneath" it, the study team dubbed this the "hover orbit." The study found that for such a mission, technologies involved with operations are approximately equal to power and propulsion technologies in importance and the need for advanced development; therefore, they were added to the list of emphasized technologies.

The study science team, composed of a Giant Planets Panel representative and two collaborators, specified that the highest-priority science observations would be done by two imaging instruments. A narrow angle camera (NAC) with resolution of 10 cm or better (1 cm goal) from the hover orbit would allow analysis of individual collisions and determination of particle spin states for the largest common particles, 1–10 m in diameter. A wide angle camera (WAC) would provide context for the NAC images and permit larger-scale surveys for aggregate behaviors such as self-gravity wakes, "propellers," etc. Lower-priority instruments were mentioned, but with one exception they were not used in this study. The exception is an instrument, possibly some variant on a laser altimeter or LIDAR (light detection and ranging), that would measure the distance to the ring particles, ring thickness, and the z-axis (perpendicular to the ring plane) components of ring particle velocities. The study found that this ability to measure the distance from the spacecraft to the ring plane was an important engineering function as well as science function; therefore, such an instrument is included in the planning payload. This is not a particularly demanding payload. Even the NAC requirements are met by imaging systems currently in flight on competed missions; thus, a payload of ~30 kg and ~30 W is sufficient, assuming that the ranging instrument is similar to a scanning LIDAR.

Another important science finding from the study is that the primary metric for mission science value is the range of radial traverse (with respect to Saturn's center) over the mission lifetime. There is a strong correlation between this radial traverse range and the number of different ring environments and known phenomena encountered, and thus the value of the science obtained. The science team identified four levels of progressively increasing science value, assuming the ring traverse begins in the gap between the A and F rings at a Saturn-centric radius of ~139,000 km: 130,000 km (level 1); 120,000 km (level 2); 100,000 km (level 3); and 84,000 km (level 4). Level 1 is considered to be the "science floor," the science value below which it would not be worth flying the mission.

The study considered a number of technologies in several different areas. For the hover orbit and the spacecraft that would operate in it (the "hover spacecraft"), radioisotope power systems (RPS) and nuclear fission reactor power systems were considered, along with propulsion technologies that included Hall thrusters, ion thrusters, colloid thrusters, arcjets, and chemical propulsion. Technologies that benefit SRO by allowing more mass to be delivered to hover orbit initiation (HOI) included solar electric propulsion (SEP), radioisotope electric propulsion (REP), nuclear electric propulsion (NEP), high-I_{SP} in-space chemical propulsion, aerocapture in Saturn's atmosphere, and aerogravity assist (AGA) in Titan's atmosphere.

For a first cut at assessing the technologies needed to achieve the different levels of science value, the study team modeled flight systems using low-fidelity, rough-order-of-magnitude (ROM) models.

Subsystems directly involved in maintaining the hover orbit were modeled at the subsystem level, but all other subsystems were modeled at the system level only, primarily by analogies to previous missions.

The level of effort needed for enabling technologies increases as the final radial distance for the mission decreases (and science return increases). Common to all missions of any science level is the need for the operations/navigation technology to fly the hover orbit—the instrumentation to reliably measure the spacecraft's distance from the mean ring plane and the autonomy to interpret those data and translate them into actions the spacecraft must take, such as increasing hover thrust, to keep the spacecraft safe and properly positioned for science data acquisition. The most scientifically ambitious missions, achieving level 4 science, would launch on a Delta IV Heavy augmented by a large SEP stage, would require the most advanced RPS systems (SRG-550), and would require either Titan AGA or high-I_{SP} in-space chemical propulsion. Fully achieving level 3 science would also require most of these technologies, but does not require the AGA or high-I_{SP} propulsion. Using a SEP-augmented Atlas V 551 and the SRG-550 would *almost* achieve level 3 science. Descoping further to level 2 science, an SEP-augmented Atlas V 551 with the advanced Stirling radioisotope generator (ASRG) currently in development would be sufficient, with one important difference from the higher levels. The lower-wet-mass spacecraft needed for the lower science levels would require smaller but efficient EP thrusters, at thrust levels and power levels where current engines are not very efficient, below 20% in some cases. Development of higher-efficiency, low-thrust EP engines would be needed to allow operation with ASRGs. If the higher-efficiency engines are not available, more advanced RPSs would be needed, such as the SRG-160 and SRG-550. At level 2 science, the ability to fly the hover orbit with chemical propulsion emerges, though the option requires either a SEP-augmented Atlas V-551 or a Delta IV Heavy launch capability. The chemical mission is very different from the electric propulsion (EP) mission, in that it would not maintain a constant offset distance, and traverse would occur in brief Hohmann-like spurts. Also, mission duration would be significantly shortened, though the total time spent in a full hover would actually be greater. This is also true for the minimal level 1 science. At level 1, the chemical system *might* fly on an Atlas V-551 without SEP, while the EP systems appear to fit easily on such a launch system, assuming the increase in low-power EP engine efficiency.

If an REP implementation of the hover-traverse capability is not available and an EP implementation is desired over a chemical implementation, there is an option for an NEP implementation. For the REP systems, level 1–3 science typically requires total power levels in the ~0.5 to 2 kWe range. Less than 5 kWe is the extreme low end of the practical range for nuclear fission electric power systems, so the specific power of such systems cannot compete even with that of the current-technology ASRG. This suggests that the pairing of NEP with an SRO mission is not as good a match as pairing REP with SRO. It could indeed be done, but at a cost of much more mass (and thus money) required to solve the problem. As mentioned above, delivering sufficient mass to HOI to achieve level 1 science with REP or chemical implementations for the hover spacecraft requires launching on an Atlas V 551, possibly with SEP augmentation. Delivering sufficient mass to HOI to achieve level 1 science with a NEP implementation for the hover spacecraft requires launching on a Delta IV Heavy with SEP augmentation, a much more expensive means of delivery to Saturn. Achieving level 2 science with NEP requires that *and* either a Titan AGA or high-I_{SP} chemical propulsion for SOI/pumpdown. At much higher power levels (and masses), well above 10 kWe, NEP might provide some scientifically interesting mission options, but most likely in many aspects they would differ significantly from the comparatively simple architectures and mission profiles examined in this study.

A surprising result from the study is that at the lower science levels, the capability to fly this mission is closer to achievable than previously thought. This is mainly due to advances in trajectory techniques (note: *not* navigation or maneuver-execution techniques, requiring more accuracy in trajectory operations) that drastically reduce mission delta-V requirements. At science levels 1 and 2, there are mission concepts that could fly with almost-ready power technology, incremental improvements propulsion technology, and the operations technology. Bringing the ASRG to flight readiness fulfills the power system need regarding EP hover propulsion. Increasing the global efficiency (jet power divided by power out of the power source, so including PPU efficiency) of low-power electric thrusters, like small Hall thrusters, to 40% or more, fulfills the propulsion need. Operations technologies, especially the ranging and navigation technologies, might require as much development effort as the power and propulsion technologies for these low science level missions.

Study Purpose and Objectives

The purpose of this study was to characterize and evaluate technology needs (particularly power and propulsion) and trajectories to support a potential future mission to observe the rings of Saturn at close range. Technology needs were identified for development within the time span of this Decadal Survey to support investigation of potential future mission concepts in the following Decadal Survey.

The principal objectives of this study were to:

1. Assess the feasibility of an SRO mission, using either presently available technology or hardware that might become available in the next decade.
2. Investigate the method(s) by which a spacecraft might be placed in a tight circular orbit, using chemical or nuclear-EP or aerocapture in Saturn's atmosphere.
3. Identify plausible instrument payload(s) for the orbiter and the sub-satellites, necessary to carry out the above goals.
4. Investigate methods by which the data collected could be returned to Earth.
5. Identify technological developments for the next decade that would enable such a mission in the post-2023 timeframe (after the next saturnian equinox).

Study Approach

This study differed from other studies supporting the 2012 Planetary Science Decadal Survey: it did not generate flight or ground system point designs for any mission concept, did not perform any detailed spacecraft subsystem analyses beyond simple parametric modeling, and did not examine a wide architectural trade space to generate a set of preferred options. Instead, the mission architecture is fairly well defined up front (the “hover orbit” and “hover spacecraft”). The study focused on potential implementation options for this architecture and on technologies needed for those options, specifically technology developments needed in the 2013–2022 timeframe to allow a flight project in the decade following that. The study questionnaire provided by the science team [1] calls out power, propulsion, and trajectory technologies as the highest priorities, though other technologies were considered as their mission impacts became clearer.

For the ring hover architecture specified, flight system requirements naturally flow backward in mission time, from the hover spacecraft that would perform the science mission back to the system launched from Earth, including the launch vehicle. In addition to the hover spacecraft, the architecture would include a Saturn orbit insertion (SOI) / pumpdown stage, and any cruise or SEP stage or extra hardware needed between launch and Saturn approach; see Section 2, Overview. For each of these flight elements, there are multiple candidate technologies that might be used to implement them. In some cases, there is overlap. For instance, EP, which could be used to perform the hover spacecraft’s hover and traverse functions (see Section 2), might also be used in the pumpdown phase or in the Earth-to-Saturn transfer. Using the backward flow of requirements, initial analyses determined the viable implementation pathways for the different levels of science (related to the distance of radial traverse, discussed in Section 1) to be achieved by the hover spacecraft and the technologies associated with those pathways. For each of those viable pathways, there are implementation options for the SOI/pumpdown stage and the technologies associated with each pathway. Further back in mission time, there are options for delivering the combined masses of the hover spacecraft and SOI/pumpdown stage to Saturn approach and their associated technologies. This trace back approach was used to guide the modeling of flight system masses to judge whether a given implementation pathway would yield a potentially feasible mission concept. For a complete set of options considered for each leg of the flight, see Figure 2-7 in Section 2, Key Trades.

Flight system feasibility was assessed using simple parametric modeling at the system level only. To model the flight system masses, the study team used a small number of parameters to describe the performance levels of power and propulsion technologies, rough bus mass and power figures from analogous missions (adjusted for such modifications as higher downlink data rates), delta-V requirements derived from trajectory analyses, and the science mission requirements. Modeling of the various implementation pathways was done over a range of requirement parameters that spanned all science levels to identify thresholds where lower-level technologies became either impractical or greatly inferior to advanced, lower-TRL technologies (see discussion of four technology levels in Section 2, Technology Maturity). A pathway was judged “impractical” if its required launch mass exceeded the launch capability of a SEP-assisted Delta IV Heavy or if a critical commodity or technology would not be available. Examples include the scarcity of plutonium for multiple RPSs or the unavailability of appropriate thermal protection system materials for aerocapture. Though not called for in the questionnaire, qualitative descriptions of risks were captured and accompany the technologies and mission concepts.

This was a collaborative study involving the Jet Propulsion Laboratory (JPL) and NASA’s Glenn Research Center (GRC). GRC provided the power and propulsion technology descriptions and trajectory analyses for Earth-to-Saturn transfers. JPL provided study leadership, science coordination with the Giant Planets Panel and other members of the science team, flight system modeling, a broad search of trajectory paths and rough flyby dates for Earth-to-Saturn trajectories, hover orbit analyses, SOI/pumpdown trajectory analyses, and leadership of the study report generation task. The study process used a combination of plenary sessions and individuals or small teams (two to three persons) working between the plenary sessions. Technical analyses were performed almost exclusively in the individual or small-team

environments. The plenary sessions were used to report and discuss intermediate and preliminary results or problems, generate new candidate options for assessment in the study, and coordinate follow-on work.

Results from the analyses provide the basis for conclusions about the technology benefits to the various implementation pathways for all four science levels. Preliminary conclusions were reported to the science team and to Leonard Dudzinski, the NASA point of contact (POC) for the study. Final conclusions, and the analyses that support them, are reported in this document.

1. Scientific Objectives

Science Background

Although much has been learned of the dynamics of planetary ring systems by remote sensing observations carried out by Earth-based facilities (chiefly stellar occultations) and by flyby and orbiting spacecraft (Voyagers 1 and 2, Galileo and Cassini), most of the fundamental interactions in rings occur on spatial scales that are unresolved by such techniques. Typical particle sizes in the rings of Saturn and Uranus are in the 1 cm–10 m range, and average interparticle spacings are a few meters. Indirect evidence indicates that the vertical thickness of the rings is as little as 5–10 m, which implies a velocity dispersion of only a few millimeters per second. In Saturn's A and B rings, the typical ring particle experiences 3–10 collisions per orbit (Period = 10 hrs, roughly), with a mean free path between collisions of 3–10 m. It is the cumulative effect of these innumerable gentle collisions that results in the outward transfer of angular momentum, which is believed to dominate all collisional disks—circumstellar or circumplanetary—and thus the gradual radial spreading of the rings. Theories of ring structure and evolution depend on the unknown characteristics of these collisions, especially the coefficient of restitution, and on the size distribution of the ring particles. At the present time, our knowledge of the former is limited to a few laboratory experiments with cold ice. The latter is fairly well-determined by spacecraft radio occultation experiments. Direct measurement of both the coefficient of restitution (by monitoring individual collisions) and the velocity dispersion of particles would provide critical data on the rings' viscosity, enabling tests of the "standard model" that underlies most theories of ring structure and evolution.

Numerical simulations of Saturn's rings incorporating both collisions and self-gravity predict that the ring particles are not uniformly distributed, but are instead gathered into elongated structures referred to as "self-gravity wakes," which are continually created and destroyed on an orbital timescale. Theory indicates that the average separation between wakes in the A ring is of order 30–100 m, a result supported by simulations. Cassini observations have produced abundant evidence for self-gravity wakes in the A and B rings from stellar and radio occultations, optical imaging, and thermal-infrared maps, while recent numerical models suggest that the wakes may also play a major role in mediating ring viscosity. Direct imaging of self-gravity wakes, including their formation and subsequent dissolution over a period of a few orbits, would provide critical validation of these models and might also shed light on many other unexplained aspects of the way in which electromagnetic radiation interacts with the rings.

High-resolution observations of individual ring particles should also permit estimates of particle spin states, which have a significant impact on the large-scale thermal behavior of the rings. Simulations suggest that the largest particles should rotate at near-synchronous rates, with smaller particles spinning faster (period ~1/r), but this is based on idealized models of spherical particles.

Finally, there is evidence in Cassini data of other organized structures in the rings at the hundred meter to kilometer scale. These include "propellers" (thought to be the signature of sub-km moonlets embedded in the rings), the "ropy" and "straw" structure seen in images of strong density waves and gap edges, chaotic structure in the outermost B ring, and curious radial oscillations observed in radio and stellar occultations of the inner A ring. The latter have been interpreted in terms of the "viscous overstabilities" seen in numerical simulations of high-optical depth rings. Again, direct observations of these structures, and their developmental time scales, would provide powerful tests of the numerical models, as well as a solid basis for interpreting remote sensing data.

Observational Requirements

Most of the science goals identified above could be accomplished by high-resolution nadir imaging of the rings from a platform that co-orbits with the ring particles; i.e., from a spacecraft in circular orbit above or within the rings. In order to resolve at least the largest (1–10 m) sized particles, a minimum spatial resolution of 10 cm is required (with 1 cm a desirable goal), corresponding to 30 microradians at a range

of 3 km. A typical 1 megapixel CCD (charge coupled device) would then have a field of view (FOV) of 100 m, sufficient to encompass one or more self-gravity wakes and more than enough to cover the mean free path between collisions. At a typical particle relative velocity of 1 mm/sec, a frame every minute should suffice to characterize individual collisions, while also providing useful data on particle spins. A wide angle camera (WAC) (FOV ~1 radian) would permit larger-scale surveys for propellers and maps of ring structures such as wakes, ring edge structures, and unstable regions.

Other orbiter instrumentation might include a laser altimeter/range-finder (to measure the effective thickness of the rings and the vertical component of particle motions), as well as a simple ultraviolet (UV) or infrared (IR) photometer to measure optical depths via stellar occultations*. In situ* instruments to measure the density and composition of the neutral and ionized ring atmosphere, meteoritic and secondary dust fluxes, and local electric fields (especially in spoke regions) would also be of great value.

Science Objectives

The Saturn Ring Observer (SRO) mission would be a high-resolution study of the microphysical interactions between particles in Saturn's rings, at a spatial scale of 1–10 centimeters and over continuous periods of order 10 hours (1 orbit), with the goals of:

1. Characterizing the coefficients of restitution, both radial and tangential, for typical collisional interactions between ring particles
2. Measuring all three components of the ring particles' velocity dispersion
3. Studying the development, dimensions, packing density, and eventual dissolution of self-gravity wakes and similar structures
4. Studying the detailed structure of "propellers," perturbed ring edges, density waves, etc.
5. Characterizing the size distribution and spin states of the larger ring particles

Ideally, the above should be done at several (at least three) representative locations in the A, B, and C rings, plus several targeted features such as gap edges and strong waves, with continuous observations carried out over time periods of several orbits (i.e., a few days) at each location.

With this in mind, the Giant Planets Panel identified four "regions of scientific interest" (science levels) and the specific properties of these regions:

1. Science floor: 137,000–130,000 km (Outer A ring)
   A ring edge, density wave region, propellers, self-gravity wakes, and Encke and Keeler gaps
2. 130,000–120,000 km
   Remainder of A ring, including optically thick inner region, outer portion of Cassini Division, including Cassini Division ringlets
3. 120,000–100,000 km
   Remainder of Cassini Division, outer B ring edge, and thickest portion of B ring core
4. 100,000–84,000 km
   Remainder of B ring, C ring plateau, and gap region

Each region has unique properties that both challenge and constrain models of ring formation and evolution. Thus, the science value increases with examination of every additional region. The spacecraft resources required to observe a region are inversely proportional to the radius. For this reason, observations of the regions of interest are identified with science levels with measurement of region 1 corresponding to science level 1.

Traversing regions 1–3 would span the full range of optical depths within the rings, from diffuse material (Cassini Division, which is similar in many respects to the C ring) to the thickest core of the B ring.

2. High-Level Mission Concept

Overview

The SRO mission concept is best understood by initially tracing it backward in time from its science orbit to its launch from Earth, through three main mission phases: the science orbit phase, also called the "hover orbit" phase; the Saturn orbit insertion (SOI) and "pumpdown" phase, beginning upon Saturn approach and ending at hover orbit initiation (HOI); and the launch and transfer to Saturn. Each phase has its own objectives that drive flight system requirements and operations strategies. Of course the primary objectives for the hover orbit phase are the science objectives, so that platform, the "hover spacecraft," must support those functions, and its mass reflects what is necessary to carry out those functions. The primary objective of the SOI/pumpdown phase is to maneuver the hover spacecraft from Saturn approach to the state (position and velocity) for HOI. This phase would involve a challenging level of propulsive delta-V, a complex tour of Saturn's largest moons for gravity assists, and significant time. The large delta-V requirement is achieved best with a dedicated SOI/pumpdown stage that performs the maneuvers for this and then is jettisoned just prior to HOI. Finally, the primary objective of the launch/transfer phase would be to deliver the combined mass of the hover spacecraft and the SOI/pumpdown stage to Saturn approach, with the approach parameters (declination and V-infinity) within acceptable limits and within an acceptable cruise duration.

The remaining mission description traces events in proper time order. This study covered many different viable architectural options, not just a single point design; thus, the description of each phase and its associated hardware options and operations covers a range of characteristics. There are limits to the characteristics ranges, however, especially when mission duration is involved. A common mission design technique is to buy greater delivered mass capability with longer cruise durations. But spacecraft components, notably radioisotope power systems (RPSs), have useful lifetimes that the prime mission duration should not exceed. For RPSs, that lifetime (from launch) is currently 14 years, assuming 3 years from fueling to launch. If the processing period between fueling and launch could be shortened to as little as one year, as has been suggested, then the post-launch useful life could be extended to 16 years. With the SOI/pumpdown phase lasting as long as 4 years, and reserving at least 1 year for the hover orbit phase, transfer trajectories requiring more than 11 years can be ruled out. This study identified mission architectures and associated trajectories that could accomplish an SRO mission within the current ASRG lifetime limits. But, extending that lifetime by one or two more years could open up more opportunities, providing added implementation flexibility, if other spacecraft components also have sufficient longevity.

A typical SRO mission option would begin with a launch and some combination of chemical propulsion, gravity assists, and electric propulsion (EP) such as solar electric propulsion (SEP), to effect the transfer to Saturn. The precise combination of launch vehicle choice and these post-launch options depends on the required mass to be delivered to Saturn approach (see Section 3, Concept of Operations and Mission Design), which is in turn a strong function of the required hover spacecraft mass, which is itself a function of the level of science the hover spacecraft addresses (see Section 1, Science Objectives).

The SOI/pumpdown phase would begin at Saturn approach. Approach targeting would lead to the SOI maneuver that could be achieved either by a Cassini-like propulsive maneuver series (formal SOI maneuver followed by a critical periapse raise maneuver [PRM]), or a Titan aerogravity assist (AGA). The all-chemical option's large SOI and PRM is followed by an approximately 3.5-year pumpdown tour would deliver the hover spacecraft to HOI. This tour would use multiple gravity assists from (in order) Titan, Rhea, Dione, and Enceladus, along with "leveraged" propulsive maneuvers (totaling approximately 350 m/s delta-V), to deliver the spacecraft and the SOI/pumpdown stage to an orbit whose periapse is just outside the F ring and whose apoapse is at Enceladus' orbit radius. A large maneuver would then circularize the orbit just outside the F ring. From there, two final maneuvers would "hop over" the F ring and circularize the orbit between the F ring and the outer edge of the A ring. The hover spacecraft would then be poised for HOI, and the SOI/pumpdown stage would be jettisoned, having provided ~3.5 km/s total delta-V, from SOI to HOI.

If instead a Titan AGA is used, the delta-V provided by that maneuver would replace the chemical option's SOI, PRM, and at least part of the pumpdown tour. The orbit state after the AGA is similar to the chemical option's orbit state after a couple of Titan gravity assists. In addition to saving significant propulsive delta-V, the AGA would save pumpdown duration as well, though at some additional mission risk. After the AGA, the pumpdown to HOI would be much the same as for the analogous parts of the all-chemical option. The AGA itself is similar to an aerocapture maneuver, but is actually less demanding in terms of heat loading. AGA maneuvers can be lift-dominated, drag-dominated, or a mix of the two. For lift-dominated AGA, the lift provided by an atmosphere's effects on an aeroshell, directed toward the primary's center, redirects the spacecraft's velocity vector by an amount larger than possible using gravity alone. In effect it uses aerodynamic lift to simulate a more massive primary. Some energy loss due to drag is unavoidable, but that is secondary to the velocity redirection. A drag-dominated AGA relies primarily on aerodynamic energy dissipation. An AGA for the SRO mission would most likely be a mix of redirection to yield the proper Saturn periapse radius, with capture into Saturn orbit provided by a net drag delta-V of about 1 km/s. That delta-V, a factor in the total heat load on the aeroshell, compares to the ~4 km/s drag delta-V for a Titan aerocapture maneuver from a Cassini-like Saturn orbit. The drag-dominated AGA also needs a lifting aeroshell, since that lift is the means of controlling the trajectory to achieve the proper exit state. During the AGA maneuver the spacecraft periapse at Titan would be in the 300-400 km range, slightly higher than needed for aerocapture. Like aerocapture, AGA has not yet been demonstrated in flight, and is subject to many of the same implementation and mission risks.

The primary objectives during the hover orbit are SRO's science objectives and the observations and measurements needed to achieve those objectives. These objectives define the functions the hover spacecraft must perform. This platform must place the payload at a safe distance (2–3 km) from the ring plane, co-orbiting with the ring particles "beneath" the platform. It must maintain that distance despite the Saturn "tidal forces" (actually a natural result of Keplerian orbital motion) that would draw the spacecraft into the ring plane. Much as a helicopter sometimes "hovers" over a point of interest on the ground, the platform could use low-thrust propulsion to maintain a non-Keplerian "hover" over a point in the rings; hence the terminology, "hover spacecraft." In addition to the hover function, the spacecraft would perform other necessary functions to acquire and return science data. It must point the science instruments toward the co-orbiting ring particles, and store and downlink (possibly after significant compression) the resulting science data. Since measurements would be needed at more than one location, it must also provide "traverse", or radial mobility across significant portions of the expanse of Saturn's rings, in the form of propulsive delta-V. Electric propulsion appears far more efficient and could enable a far greater science return than chemical propulsion, which could provide very limited capability for hovering and traverse.

To establish the hover orbit from the circular orbit between the A and F rings, the hover spacecraft would begin thrusting with its EP system perpendicular to the ring plane, pushing it to its non-Keplerian position a few kilometers out of the ring plane, on the sunlit side of the rings. Then, either by starting an additional set of EP engines or by gimbaling the already-operating engines and increasing their thrust, the hover spacecraft would provide an additional component of thrust in the direction opposite its orbital velocity vector, the "traverse propulsion." This would decrease its orbital energy, thus decreasing its orbit radius, moving toward Saturn and over the A ring. When its orbit radius reaches a point where detailed science observations are desired, the traverse propulsion would shut off and the hover spacecraft would remain hovering over the same small group of ring particles, imaging their shapes, motions, and clumping behavior until sufficient data are acquired; traverse propulsion would then resume.

The science team described a limited set of potential hazards the hover spacecraft might encounter. In some cases, especially near ring and gap edges shepherded by small moons, ring particles are perturbed up to 4 km out of the usual ring plane. Some individual objects ("ring moons") are so large that they extend kilometers out of the ring plane. The hover spacecraft must "hop over" such hazards. Fortunately, a relatively small amount of chemical propellant and small thrusters could provide this capability. Appendix C discusses known hazards in Saturn's rings.

Typical durations for the hover orbit would be one to two Earth years for EP hover implementations, considerably less for chemical implementations. During system analysis (see Section 2), it was found that there is a mass-optimal mission duration, which implies that a mission can be too long, or *too short*. The mission would end when the EP propellant is exhausted and the spacecraft sinks into the ring plane to become yet another ring particle; see Section 3, Planetary Protection. Having the spacecraft co-orbiting

with the ring particles within the ring plane is not a part of the SRO operations concept due to risks associated with collisions with the larger ring particles and the negative impact on science observations by perturbing the natural collisions of the particles.

This study did not consider mission science enhancements such as free-flying sensors that would be released into the ring plane for radio tracking, yielding such information as collision frequencies and typical random velocity magnitudes. Such sensors, if relatively short-lived (a few Saturn orbits or less), would not be particularly difficult to implement. Future studies could better define the science value to be gained from them, and might estimate mass and power levels needed.

Concept Maturity Level

Table 2-1 summarizes the NASA definitions for concept maturity levels (CMLs). The objective of this study was to identify and evaluate, at a CML of approximately 2, technologies needed for SRO mission concepts, especially technologies needing development in the 2013–2022 timeframe for a flight mission in the following decade. The study team included individuals representing the science objectives (from the Decadal Survey Giant Planets Panel) and payload capability, and overall mission, technology, and system expertise. To satisfy the range of science objectives, mission concept feasibility was assessed across a wide range of trajectories and power and propulsion technologies. Impacts of the technology options were assessed for feasibility and relative science performance, as judged by the radial extent of the ring system that could be covered. Since technical feasibility based on alternative technology options was the goal of this study, no quantitative cost analysis was performed. The results of this study provide valuable information for potential follow-on trade space or point design studies.

Table 2-1. Concept Maturity Level Definitions

Concept Maturity Level	Definition	Attributes
CML 6	Final Implementation Concept	Requirements trace and schedule to subsystem level, grassroots cost, V&V approach for key areas
CML 5	Initial Implementation Concept	Detailed science traceability, defined relationships, and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy
CML 4	Preferred Design Point	Point design to subsystem level mass, power, performance, cost, risk
CML 3	Trade Space	Architectures and objectives trade space evaluated for cost, risk, performance
CML 2	Initial Feasibility	Physics works, ballpark mass and cost
CML 1	Cocktail Napkin	Defined objectives and approaches, basic architecture concept

Technology Maturity

This study considered several technologies in two general categories: power and propulsion, and operations. Power and propulsion technologies covered a variety of technology readiness levels (TRLs), from currently flying Hall thrusters to small nuclear fission reactor power systems at a much lower TRL. Two different parts of an SRO mission, the Earth-to-Saturn transfer phase and the near-Saturn phases, would utilize different power and propulsion technologies. Operations technologies primarily focused on the technologies needed to navigate and control the hover orbit, and include sensor technologies needed to reliably detect the distance to the ring plane together with the algorithms to allow the spacecraft to autonomously control its position based on the sensor input. Currently, both areas are at low- to mid-TRLs.

The Earth-to-Saturn transfer required no new enabling technologies, but SEP proves to be highly beneficial in some cases. Notably, it would be useful for launch and transfer to Saturn in years when a Jupiter gravity assist (JGA) is not available, greatly increasing the mass a given launch vehicle could deliver to Saturn approach without the JGA. SEP is already demonstrated in space, with NASA's Deep

Space 1 (DS-1) mission and the currently flying Dawn mission, and the European Space Agency's (ESA's) SMART-1 mission. Systems that operate only one thruster at a time are at TRL 6 or higher. SEP systems of the size (i.e., power and propellant load) and complexity needed for an SRO application have not yet flown and are thus at a lower TRL, but have been the subject of multiple studies, including the recent Titan Saturn System Mission (TSSM) study [2].

Power and propulsion technologies for the near-Saturn phases were organized into four levels. Technologies in the first level include ones needing minimal or no further development, such as advanced Stirling radioisotope generators (ASRGs) and sub-kW Hall thrusters, and standard chemical propulsion systems. TRLs for these technologies are at 6 or higher. The second level introduces more extensive developments and include the "optimized" ASRG (aka the SRG-160) and the larger SRG-550, both with lower TRLs in the 4–5 range. Aero-assist technologies enter at the third level with Saturn aerocapture and Titan AGA into Saturn orbit. Aero-assist introduces more complexity into the topic of technology maturity, separating maturity of component technologies from maturity of the system technology. Aerocapture and AGA are system technologies, dependent on many component technologies such as aeroshell geometries; thermal protection system materials; aerothermodynamic modeling; guidance, navigation, and control (GNC) algorithms; and sensors and actuators for GNC. TRLs for all of these technologies are high, at 6 or above, for Titan AGA. For Saturn aerocapture, some of these components, especially aeroshell geometries, thermal protection system materials, and aerothermodynamic modeling, are at lower TRLs. TRLs for aerocapture at the system level are currently a topic of debate in the community. Some insist that single-pass, high-delta-V aero-assist techniques such as aerocapture and AGA cannot be used without a flight demonstration. Many experts insist that at some destinations, especially Titan, these techniques are ready for use now. The fourth technology level includes small (1–3 kWe) nuclear fission reactor power systems, with component TRLs ranging from 3–9, and a system level TRL at approximately 3.

The study science team identified four levels of science value that have implications for the power and propulsion technology levels. One study result is that the science value of an SRO mission increases with the range of radial traverse accomplished. Thus, for a given payload, technologies that provide greater traverse capability yield greater science value. Assuming HOI at ~139,000 km Saturn-centric radius (in the clear zone between the F ring and the outer edge of the A ring), the team identified four threshold radii for progressively higher science value levels. Moving from one science level to the next higher generally requires adopting more aggressive technologies and, presumably, increased risk to meet the increasing delta V requirements for traverse of the rings.

A somewhat surprising finding from the study is that (assuming the availability of operations technologies needed for the ring traverse) even with level 1 power and propulsion technologies, there appear to be feasible missions using launch vehicles significantly smaller than the Delta IV Heavy. *Thus, the operations technologies could impose more of a constraint on mission readiness than the power and propulsion technologies.*

Key Trades

This study was intended to investigate a broad swath of the technological possibilities to identify technologies needed to enable an SRO mission. Multiple techniques were employed to investigate the impact of technology both on the core hover spacecraft and on the rest of the launch stack responsible for delivering the hover spacecraft to HOI, the location, and velocity at which the hover spacecraft begins its non-Keplerian hover orbit.

The SRO study presented both discrete and continuous dimensions in its trade space. Continuous parameters described the mission and the technology metrics of the hover spacecraft. Discrete parameters were used for the choices of launch opportunities, available launch vehicles, and technologies such as SEP and Titan AGA.

The goal of investigating the mission architecture trade space was to identify which technologies could potentially be enabling for Saturn ring missions of various levels of science investigation.

Hover Design Space Exploration

There were two goals for exploring a continuous space of the hover spacecraft design parameters. The first of these was to evaluate what levels of performance would be required of EP thrusters and their onboard power source to achieve various science missions. The second goal was to evaluate which of the mission science levels would be achievable by the family of concepts considered within this study. These goals guided the exploration of hover spacecraft sizing parameters.

Examination of the hover spacecraft trade space was initiated with the selection of the input variables of interest. There were successive iterations on the hover spacecraft analysis, but each iteration included the variables listed in Table 2-2. All analyses used the variables in this table. The continuous trade space exploration (shown in contour plots later in this section) used values within the minimum/maximum values in the last two columns of Table 2-3. Separate sets of values were used for reactor- and radioisotope-based power systems. Similar but smaller sets of variables were used for analyses of hover and traverse by means of chemical propulsion; those analyses used discrete examples rather than continuous trade space exploration. For those analyses, the spacecraft bus (with instruments) was assumed to require 350 We provided by ASRGs. Chemical propulsion systems for hover and traverse used typical current state-of-the-art values for inert masses, I_{SP}, etc.

Note that another parameter, "initial radius," the radius at which the hover orbit begins, did not vary from option to option and thus was considered a constant in the analyses. Each of these parameters was assigned a range. For the final version of the continuous space exploration, which is represented in the contour plots shown later in this report, the parameters and ranges used are listed in Table 2-3. For further information on the specific assumptions of the hover spacecraft, see Section 3, Flight System.

Table 2-2. Design Space Variable Description

Variable Name	Description
Final radius (km)	The end point for a traverse radially inward over the rings
Specific power (W/kg)	The output power of the main power source divided by its mass
Thruster efficiency (%)	Jet power (mass flow multiplied by effective exhaust velocity, divided by 2) divided by the amount of power output from the power source
Payload mass (kg)	The mass of cameras and LIDAR (for altitude detection)
Mission duration (months)	Time from start of the hover orbit until the end of the mission; assumes a steady reduction in orbit radius over time
Hover duration (months)	Time allocated to stop traverse and hover at a fixed radius over the rings
Hop delta-V (m/s)	Amount of monopropellant delta-V allocated for clearing hazards encountered along the rings

Table 2-3. Design Space Parameter with Minimal and Maximal Values

Name	Comment	Minimum (value for NEP variant in parentheses)	Maximum (value for NEP variant in parentheses)
Final radius (km)	The end point for a traverse radially inward over the rings	80,000	130,000
Specific power (W/kg)	The output power of the main power source divided by its mass	5 (2)	11 (4.5)
Thruster efficiency (%)	Jet power (mass flow multiplied by effective exhaust velocity, divided by 2) divided by the amount of power output from the power source	30	80
Payload mass (kg)	The mass of cameras and LIDAR (for altitude detection)	20	80
Mission duration (months)	Time from start of the hover orbit until the end of the mission; assumes a steady reduction in orbit radius over time	6	24
Hover duration (months)	Time allocated to stop traverse and hover at a fixed radius over the rings	1	6-
Spacecraft bus power (W)	Power needed to support spacecraft bus, including instrument power	200	400
Specific impulse (s)	Design specific impulse; used together with required thrust and efficiency to calculate input power to thruster	900	1,600

A vector with values between the minimum and maximum of each of these parameters forms a design case. Six hundred design cases were run through analytical models, with the set of values within each design case chosen according to a Latin hypercube scheme. The Latin hypercube sampling technique was chosen in order to get a well-randomized draw of possible values within the design space. It was chosen because the location of break-points in the design space were not well known, and also because it was known that required delta-V for the ring mission and hover spacecraft masses would be highly variable. The results were fit using an artificial neural network. The network was used to render a series of dynamic curves and contours. The use of dynamism enabled examination of the highly dimensional surfaces within the limitations of two- and three-dimensional displays and human cognition [3, 4]. The design team used the contours to probe trends and interactions within the hover spacecraft concept and make judgments about appropriate technology strategies. Some representative views from this exploration are presented in Figures 2-1 through 2-5 to highlight the source of recommendations.

The view shown in Figure 2-1 depicts the calculated traverse range of the hover spacecraft. This captures results with input values shown in Table 2-4.

The 800-kg hover spacecraft initial mass represents the maximum HOI delivery mass for a flight system with all-chemical cruise and SOI/pumpdown stages, launching on an Atlas V 551. The shaded region in Figure 2-1 corresponds to final radii greater than 120,000 km, i.e., not fully achieving level 2 science. Figure 2-1 shows that this level of traverse is possible within the first technology level described in Section 2, Technology Maturity. This involves the ASRG with a specific power of 5.6 W / kg and a propulsion system efficiency of 40%. As shown in Figure 2-2, level 1 science is easily reached with this power system.

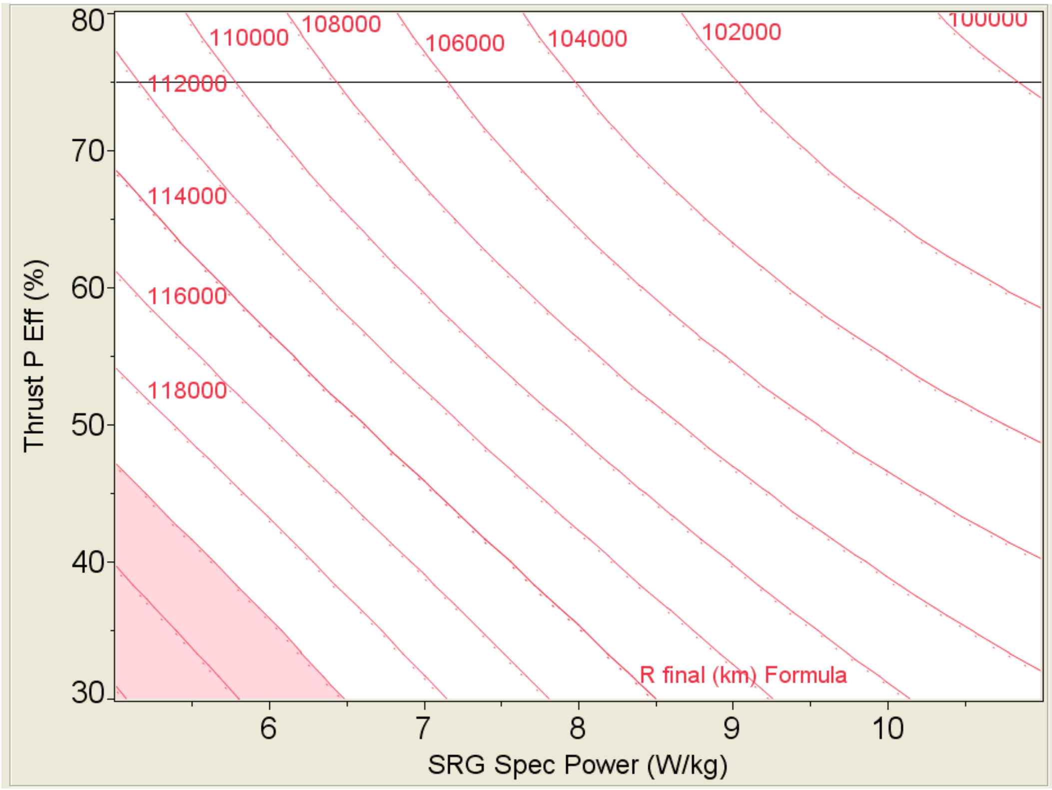

Figure 2-1. Hover Spacecraft Traverse Capability, Mass at HOI Constrained to 800 kg; Contours are Radius at End of Traverse

Table 2-4. Parameter Values for Figure 2-1

Payload Mass (kg)	Mission Duration (months)	Hover Duration (months)	SC Bus Power (W)	Specific Impulse(s)	Hover Spacecraft Mass @ HOI (kg)
30	24	2	350	1,500	800

Figure 2-2 presents an expanded view of Figure 2-1 with a wider span of values for specific power of the power source and propulsion system efficiency. The area in pink now represents the traverse constraint posed by level 1 science. It can be seen that the ASRG power technology could accommodate the mission with a low efficiency propulsion system (25% efficiency from SRG outlet to jet power).

Additional assessments were made by examining multiple two-variable plots composed of pairs of the input variables in Table 2-4 above. Other outputs were examined, such as required thrust and total power, in order to judge which technologies were appropriate for the hover spacecraft. Multiple hover spacecraft mass constraints were also examined to judge the impact of different launch vehicles and cruise stage technologies.

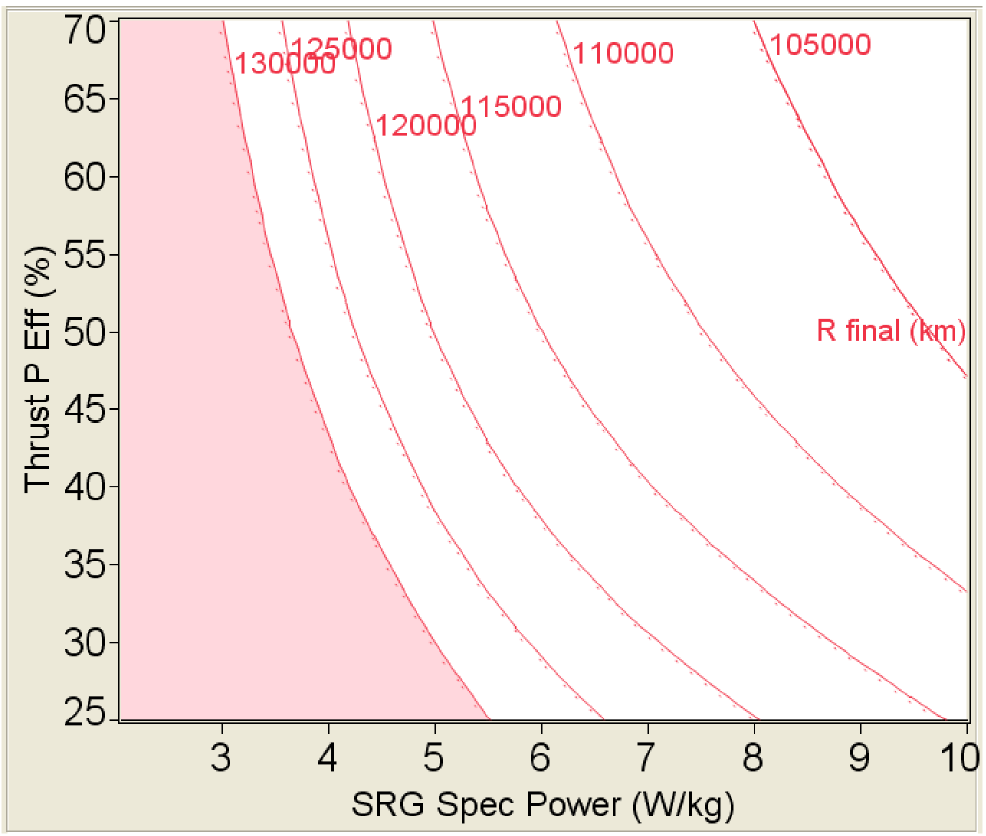

Figure 2-2. Expanded View of Figure 2-1 Showing Level 1 Science Constraint

Figure 2-3 shows a case similar to that in Figure 2-1. The main difference is in the available mass, which is increased to 1,800 kg due to the use of a Delta IV Heavy to launch the flight system stack on a SEP-assisted transfer trajectory to Saturn. The total power and required thrust contours are provided on this chart for the benefit of technologists. For SRGs, the specific power increases with the size of individual units (see the difference between the ASRG and SRG-550 in Section 3). Similarly, thruster efficiency tends to increase with thrust for a given specific impulse level. Efficiencies of 50–60% are feasible for either ion or Hall thrusters in the thrust range indicated in Figure 2-3. The SRG-550 concept that the Glenn Research Center team identified has an estimated specific power of 8.5–9 W/kg while generating ~500 W at end of mission (EOM). This unit size would be most appropriate for the higher-power hover spacecraft, such as the one indicated in Figure 2-3 with a power requirement between 2.5 and 3 kW. These values (for these technology metrics) indicate that level 4 science would require both the SRG-550 (9 W/kg) and roughly 65–70% thruster efficiency to bring the hover spacecraft wet mass below 1,800 kg.

The development of a 600 W Hall thruster with a Thrust-to-Power (T/P) level greater than 100 mN/KW and a thrust efficiency of approximately 60% would require investigating new discharge channel, magnetic circuit, and anode configurations and materials. The new thruster configuration would result in a 600 W Hall thruster with minimal losses in the discharge channel and with efficient operation at low discharge voltages (to get high T/P). To achieve the desired propellant throughput with minimal risk for the SRO mission would also require adaptation of an innovative life-extending discharge channel replacement mechanism that is currently being integrated into a flight-like 3.5 kW NASA Hall thruster. The new 600 W Hall thruster design could use existing highly efficient power processing unit (PPU) topologies and existing light-weight xenon propellant feed systems. It is projected that the desired performance levels can be achieved in 3-5 years assuming a steady level of investment is directed towards such activity. Given the performance levels and relevant physics governing the operation of state-of-art Hall thrusters at low power, achieving thrust efficiencies higher than 60% might not be possible.

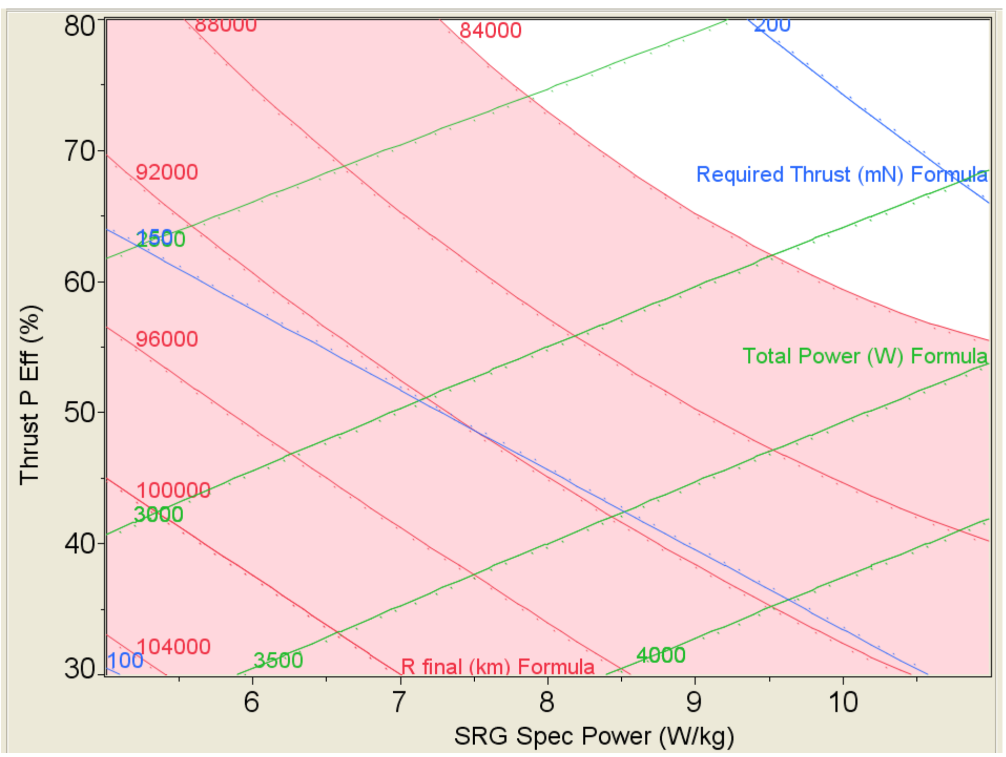

Figure 2-3. Traverse Capability for Hover Spacecraft with Multiple Outputs Displayed, 1,800 kg Constraint on Mass at HOI

To develop a higher power (100-300 W), high efficiency colloid thruster would require a robust and sustained research/development effort for the next 5 to 10 years. Colloid thrusters have been demonstrated for power levels up to 10 W. Preliminary research efforts and evaluation into scaling to higher power indicates that a colloid thruster system efficiency of 65 to 75% is attainable for power levels above 100 W. To attain such performance levels would require advancements in scaling of emitters, demonstration of new materials for long duration operation, development of high efficiency PPU designs, development of a higher-current propellantless neutralizer, and scaling the ionic liquid propellant feed system to accommodate higher power operation.

Figure 2-4 shows what is needed to achieve level 4 science when both SEP and Titan AGA would be used to bring the hover spacecraft to HOI. If the SRG-550 power source specific power value is found on this plot, then a propulsion system efficiency (including a power processing unit [PPU]) of 60–65% is sufficient to reach this science level. This represents the most capable hover spacecraft and delivery system combination in the trade space considered for this study.

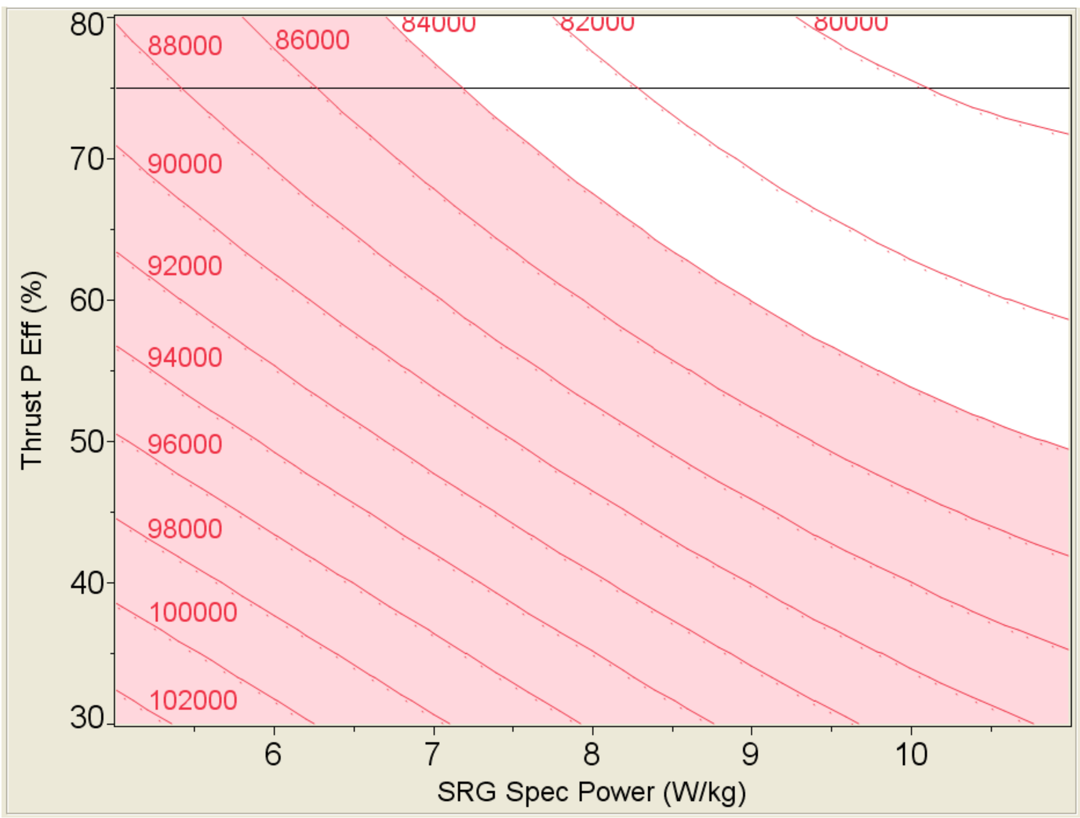

Figure 2-4. Traverse Capability for Hover Spacecraft with 2,100 kg Wet Mass at HOI

Figure 2-5 shows the results of investigating reactor-based nuclear electric propulsion (NEP). These results correspond to the input parameter settings found in Table 2-5.

The provided hover spacecraft mass corresponds to delivery by SEP launched on a Delta IV Heavy. This shows that a NEP-propelled hover spacecraft could feasibly perform a science mission that takes one year to traverse from 139,000 km to 120,000 km radially over the rings. If the total power contours (in blue) are considered along with reactor performance given in Table 3-8, it can be seen that only 1.5–2 kW reactors are feasible in this graph. Beyond that point, the specific power required to keep the HOI mass within the delivery limits of the Delta IV Heavy launch vehicle increases more rapidly than the actual specific power of a given size of reactor. The contour plot also shows that it takes a great deal more support technology and a larger launch vehicle to make the mission feasible. Low specific powers do not meet the requirement that the power source must generate enough power to “hold itself up” at an appropriate EP thrust level, with enough margin to also power the hover spacecraft. The power range appropriate to SRO is at the low end of the practical reactor system power range. There, the specific power of currently envisioned reactor systems is quite low, and significantly higher specific powers are available only from much larger systems. Thus, the scale of SRO is not a good match to nuclear fission reactor power systems.

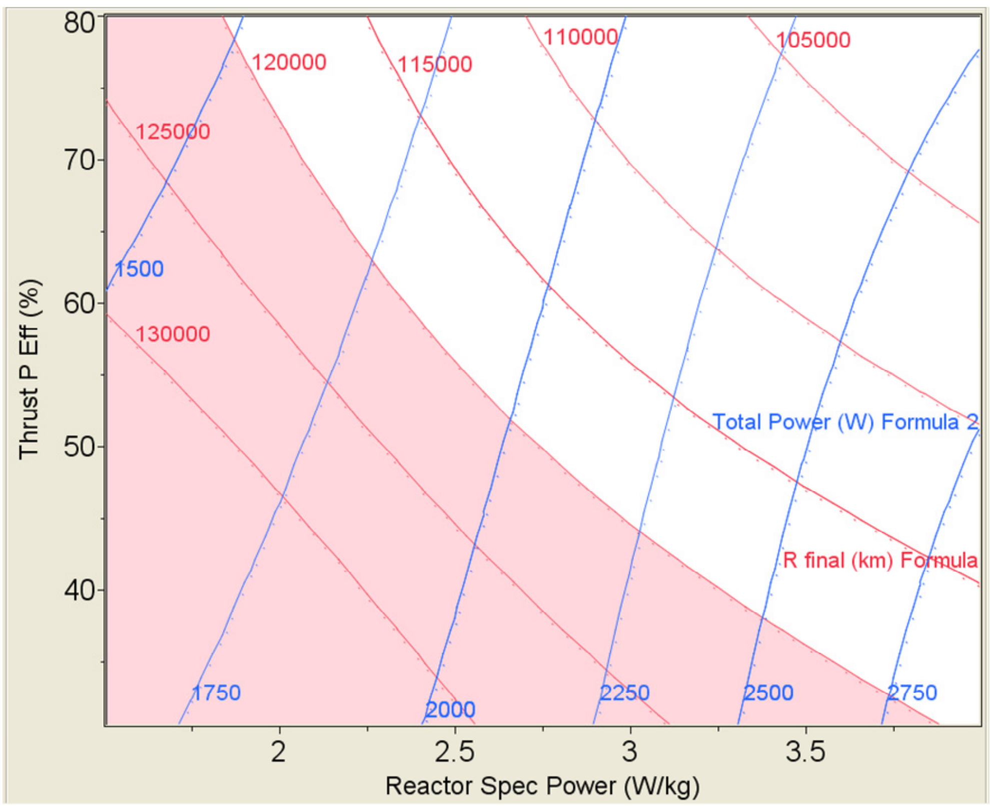

Figure 2-5. Reactor-Based NEP Trades on Mission

Table 2-5. Parameter Values for Figure 2-5

Payload Mass (kg)	Mission Duration (months)	Hover Duration (months)	SC Bus Power (W)	Specific Impulse(s)	Hover Spacecraft Mass @ HOI (kg)
30	12	1	350	1,600	1,800

Figure 2-6 compares reactor-based electric propulsion and Stirling radioisotope-based electric propulsion. This is an integrated analysis of what mass requirements various final radii impose upon the hover spacecraft, given a start at 139,000 km. Table 2-6 provides the set of values for which this plot was generated.

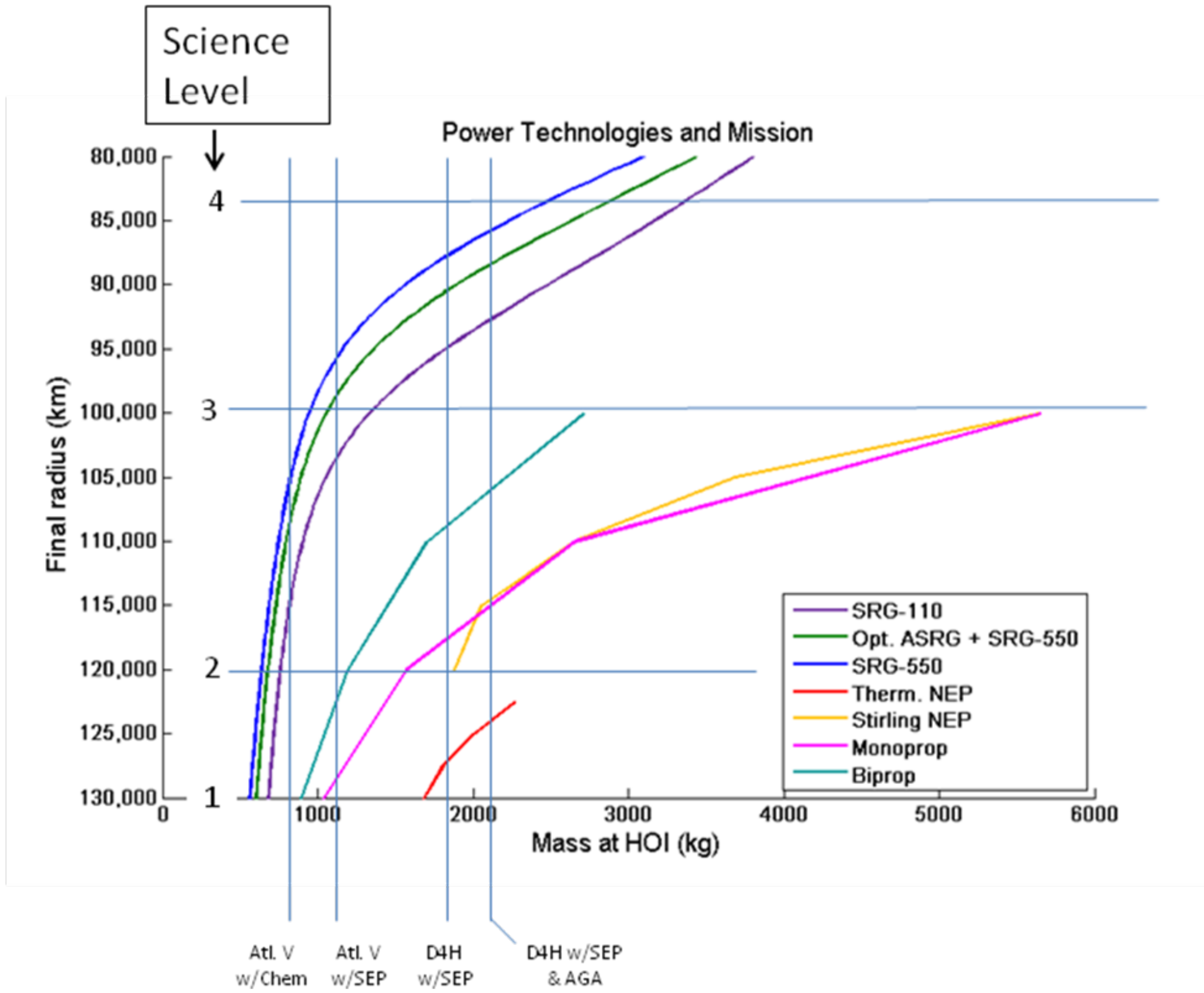

Figure 2-6. Traverse Final Radius versus HOI Hover Spacecraft Mass

Table 2-6. Parameter Values for Figure 2-6

Propulsion	Payload Mass (kg)	Mission Duration (months)	Hover Duration (months)	SC Bus Power (W)	Specific Impulse(s)	Propulsion Efficiency (%)
NEP	30	12	1	350	2,000	50
REP	30	24	2	350	1,500	50

As described by the various plots in this section, wide ranges of technology metrics and mission parameters were explored. To give a rough idea of the required total powers and thrusts for different science levels and hover spacecraft masses at HOI, the following table is presented. Table 2-7 is presented along with feasible science levels in order to give the reader a concise impression of the interplay between hover spacecraft sizing and science level.

Table 2-7. Required total power and thrust for REP hover stages of different masses at HOI; parameter values from Table 2-6 used with 8 W/kg specific power

Hover Spacecraft Mass at HOI (kg)	Required Total Power (kW, w/ 350 W for Spacecraft)	Required Total Thrust (Vectored, mN)	Feasible Science Levels
800	1.0	44	1, 2
1,000	1.5	68	2
1,100	1.7	80	2
1,400	2.3	110	3
1,700	2.9	150	3
1,800	3.2	160	4, but with higher efficiency and specific power values than used elsewhere in this table
2,100	3.8	190	4, but with higher efficiency and specific power values than used elsewhere in this table

Delivery from Saturn Approach to HOI: Trade Space Exploration

In the exploration of the discrete trade space, the study team examined a large number of possible combinations of science mission, hover spacecraft power and propulsion technology values (e.g., specific power), and transfer propulsion technologies. A variety of ways are available to meet the different levels of ambition in science missions.

Figure 2-7 shows the different elements of mission definition and technology examined in this study. The primary focus of this trade space is in propulsive technologies to deliver a hover spacecraft of the necessary mass to achieve different science missions. This set of options was used to guide investigations into appropriate trajectories between Earth, Saturn, and the rings by placing constraints on the types of propulsion to be considered. For example, NEP and radioisotope electric propulsion (REP) systems were considered with power levels under 3 kW for their effectiveness in reducing either flight time or total propellant mass. The investigations into this trade space took place in parallel with the initial investigations into the hover spacecraft. Once both of these were completed, a new combination space was developed for direct analysis. This space consists of the four different science options, seven candidate trajectories, and multiple levels of technology metrics to be used on the hover spacecraft. For this examination, only the REP version of the hover spacecraft was considered. The REP options are presented in Figure 2-8. Six hundred and sixty design cases were chosen from this space. Analysis of the results of these design cases allows for the various science mission and technology options to be traded against each other simultaneously by using the “design by shopping” technique [5]. The design by shopping technique is so-called because the options to be explored are analyzed all at once before design choices are to be made. The advantage of this method is that the options can be examined closely and in the context of each other across a large number of different dimensions. This allows those making design choices to simultaneously choose solutions and understand which dimensions are critical. Although only the REP version of the hover spacecraft was analyzed this way, the results that relate cruise stage and launch vehicle capabilities to mass at HOI are also relevant for the chemical and NEP versions.

Figure 2-7. Options for Arriving at and Performing Hover Mission

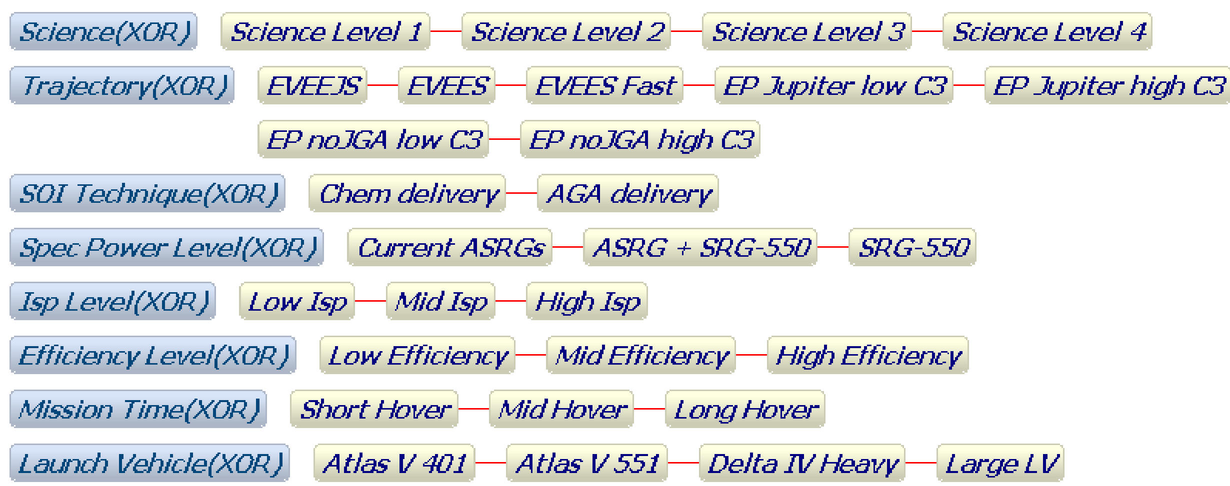

Figure 2-8. Dimensions of Combinatorial Design Space

Figure 2-9 shows results from an example of the technique. In this case, only the options with “moderate” hover spacecraft technology are shown. The points are color-coded by launch vehicle and coded by symbol for the appropriate transfer propulsion technologies. With the hover spacecraft technology metrics set, there are still many remaining dimensions in the trade space that are available for choices to be made. One major slice is shown in Figure 2-9, the tradeoff between mission time and trajectory. This plot also shows only points satisfying the level 3 science. The main point of this is to understand the effect of having a JGA available for the mission. The red line at 16 years corresponds to the assumed maximum life of a Stirling or reactor power unit.

The coding of the symbols in Figure 2-9 is as follows. Blue and green points can be launched by an Atlas V 551 with required margin. Orange points correspond to Delta IV Heavy launches. Downward facing carats represent transfers and SOI/pumpdown using chemical propulsion only, squares use Titan AGA, triangles use SEP, and diamonds use both SEP and Titan AGA to maximize delivered mass. Looking at the preponderance of color in the chart shows which trajectories tend to lead to larger launch vehicles, all other things being equal. The preponderance of blues in the “EP noJGA low C3” indicates that this is a trajectory that is less demanding than the other EP trajectories, despite being a little bit slower. Also, it should be noted that the EVEES trajectory has only one Atlas V 551 solution, with the others exceeding the 16-year assumed life of the power units.

Figure 2-10 shows how the total stack mass and ring traverse range interact with one another. This helps to illustrate directly what kind of launch vehicles and transfer propulsion are required to reach a given traverse when hover spacecraft technologies are assumed. This shows either level 2 or 3 science (final radius at 120,000 km or 100,000 km respectively) as being the most feasible with the “moderate” level of technology settings for the hover stage. Considering this information in conjunction with Figure 2-3 shows the approach toward the kinds of advances needed in the next decade to enable an ambitious SRO mission in the timeframe for the next Decadal Survey.

It is definitely noteworthy in Figure 2-10 that fixing the technology levels for the hover spacecraft leaves only the EP system specific impulse, mission duration, and traverse range as driving parameters for hover spacecraft mass. This is the reason that the results cluster on a small number of discrete hover spacecraft mass values, when the hover spacecraft mass is a continuous parameter. Furthermore, interrogating the points represented in this chart shows that there are no viable (margin >30%) missions with moderate level hover spacecraft technology with a final radius of 84,000 km. To achieve level 4 science (84,000 km) requires the higher values for specific power and efficiency than the “moderate” settings of 7.6 W/kg and 55%, as discussed in the hover spacecraft subsection. The difference between the two columns of points on the right of the chart is in the specific impulse.

The trades presented in this section are only a small cross-section of the data sets examined. The “design by shopping” technique illustrated in Figures 2-9 and 2-10 were used to make conclusions about the technological demands of different levels of ambition in science mission at the rings.

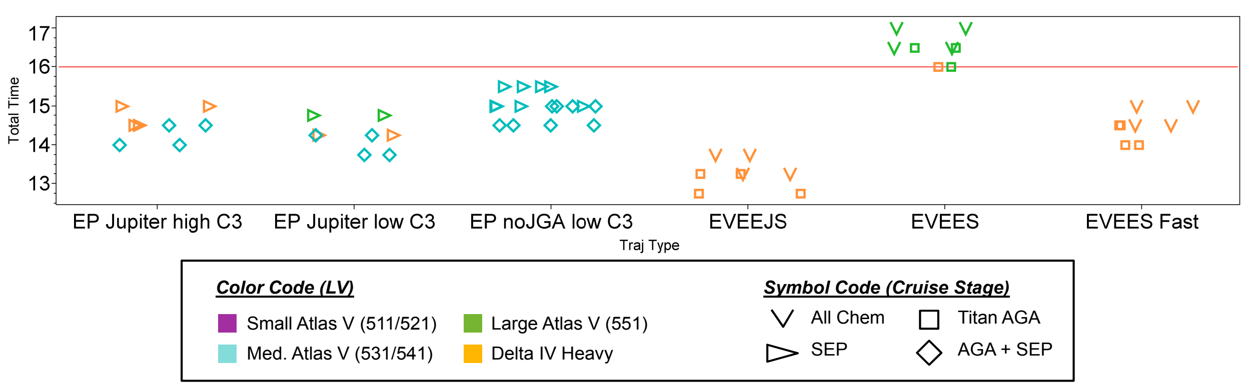

Figure 2-9. Trajectory Versus Mission Time, Moderate Hover Spacecraft Technology, Level 3 Science (see text for symbol definitions)

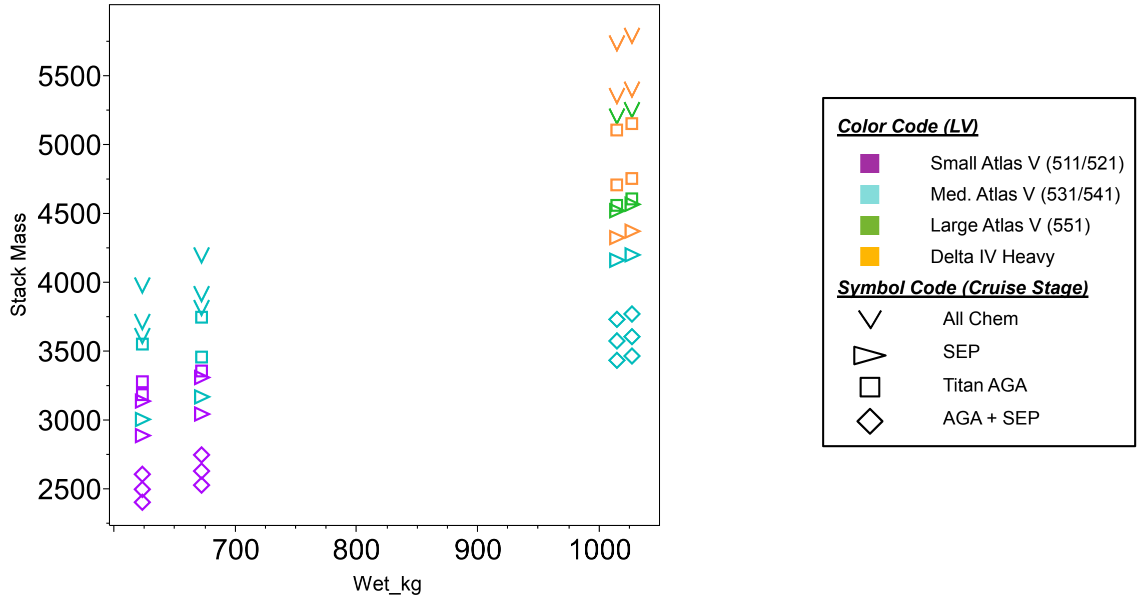

Figure 2-10. Hover Spacecraft Wet Mass and Total Stack Mass, Moderate Hover Spacecraft Technology, Two Year Mission, Science Levels 1–3

The amount of mass that different cruise stages could deliver to HOI determines the mass allocation for the hover spacecraft. For the calculations and discussion of this section, a 30% margin is maintained between delivered mass and allowed hover spacecraft mass. Table 2-8 gives the connection between the hover spacecraft masses and cruise stage characteristics.

Table 2-8. Launch Stack Characteristics to Deliver Various Masses to HOI

Hover Spacecraft Mass at HOI (kg)	Launch Stack Characteristics
800	Atlas V 551; chemical stage from Earth to end of Saturn pumpdown; chemical stage for delivery to near-HOI orbit and HOI
1,000	Atlas V 551; Titan aerogravity assist; chemical stage for delivery to HOI
1,100	Atlas V 551; SEP from Earth to Saturn; chemical stage for SOI, pumpdown, and HOI delivery
1,400	Delta IV Heavy; chemical stage from Earth to end of Saturn pumpdown; chemical stage for delivery to near-HOI orbit and HOI
1,700	Delta IV Heavy; Titan aerogravity assist; chemical stage for delivery to HOI
1,800	Delta IV Heavy; SEP from Earth to Saturn; chemical stage for SOI, pumpdown, and HOI delivery
2,100	Delta IV Heavy; SEP from Earth to Saturn; aerogravity assist at Titan; chemical stage for pumpdown and HOI delivery

The previous results give some general examples of how the discrete space was examined. To support conclusions regarding useful technologies for both the hover spacecraft and the cruise stages, the design by shopping technique was enhanced by screening away cases based on dimensions not represented in a given plot. In this case, the level of propulsion and power technology was screened based on three discrete settings for each, as identified in Figure 2-8. The specific power levels are 5.3, 7.8, and 8.6 W/kg. The propulsion efficiency levels are 40%, 55%, and 70%.

A 3x3 chart matrix of the results of this filtering is shown in Figure 2-11 (the matrix). This shows the effect of hover stage technology on the feasible space. It should be noted that science levels 1 and 2 were not examined in conjunction with high levels of power or propulsion technology, since it was determined that they would easily be viable at these technology values. This matrix was instrumental in developing the conclusions at the end of this section, as well as Table 2-7. It allows the reader to “see” into four dimensions that needed consideration to understand how hover spacecraft and launch stack technologies enable the different levels of science.

In each cell of the matrix is a graph akin to the other two that show the design by shopping technique. The architecture names correspond to launch stack descriptions. “Full Flight Group” corresponds to SEP (right triangle), the “Chem Out AGA” uses a Titan AGA (square), “Chem Out” is a chemical-only transfer (down arrow), and “AGA” uses both SEP and AGA (diamond). The coding of the symbols in Figure 2-11 is the same as for Figures 2-9 and 2-10. Purple points can be launched by an Atlas V 511 or 521. Blue and green points can be launched by an Atlas V 551. Orange points correspond to Delta IV Heavy launches. Downward facing carats represent transfers and SOI/pumpdown using chemical propulsion only, squares use Titan AGA, triangles use SEP, and diamonds use both SEP and Titan AGA to maximize delivered mass. Looking at the preponderance of color in the chart shows which mission architectures and science levels tend to lead to larger launch vehicles. For example, the middle column of the matrix can be seen to have the preponderance of color in the “L3” part of each graph shift from orange (Delta IV Heavy) to blue, green, and even purple for aerogravity-assisted SEP launch stacks.

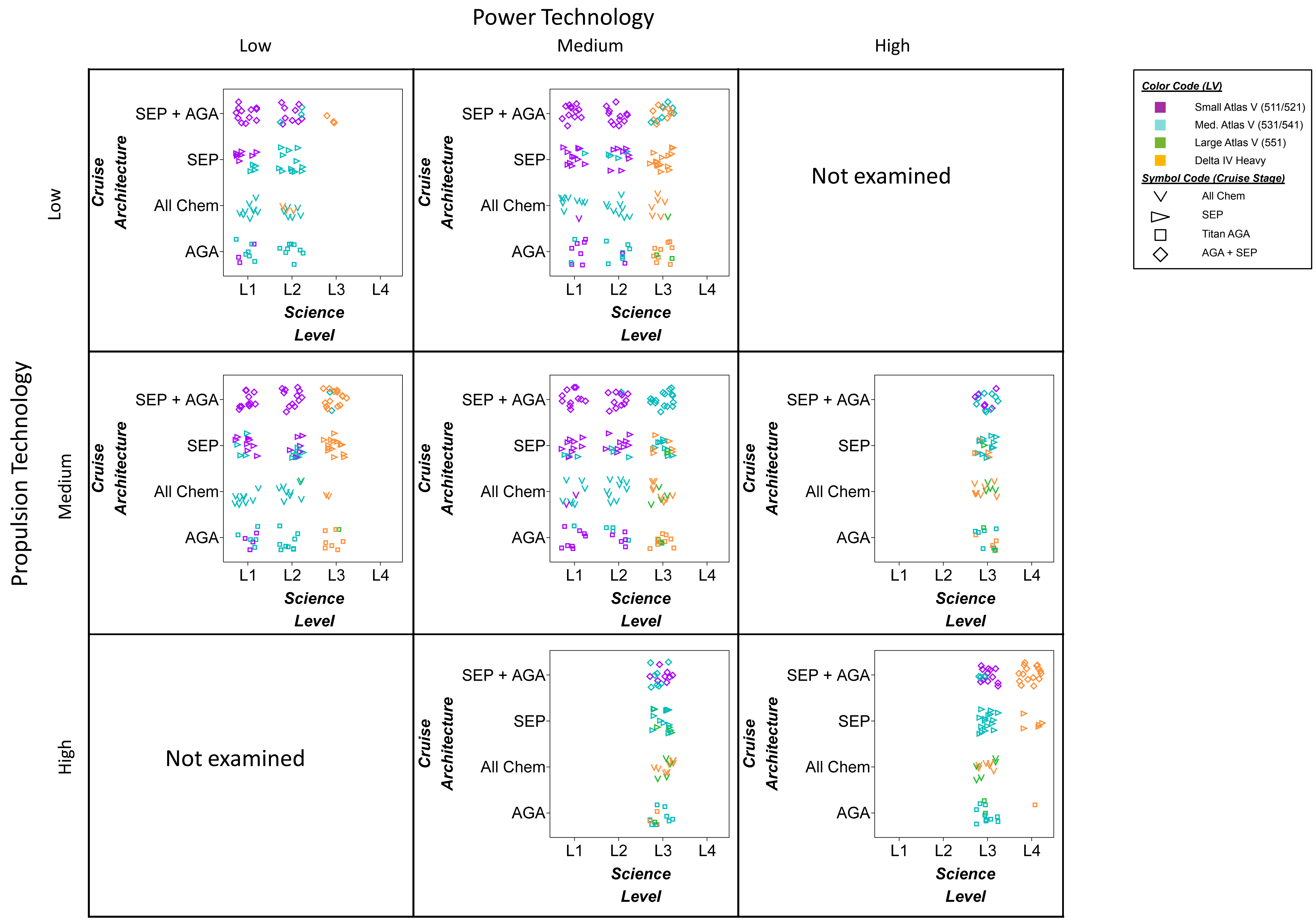

Figure 2-11. Filtered discrete space results, which are used to determine what launch vehicle and cruise stage elements would be needed to achieve different science levels.

A series of technology conclusions supported by this work are provided below.

- Deep traverse of Saturn's ring system, including over-flight of A ring, Cassini Division, and B and C rings, has many required technologies. A large Stirling radioisotope unit (550 W) plus high efficiency (70% SRG connection to jet power) electric thrusters must be developed. Also, transfer stage technology, either a large (~15 kW) SEP stage or AGA at Titan, must be used.
- Traverse of the ring system to the B ring has fewer demands on technology, but still would require investment in a large SRG unit and either: a thruster power efficiency increase (55% SRG connection to jet power), SEP, or AGA at Titan.
- A ring and Cassini division science (traversal to 130,000 km and 120,000 km, respectively) is enabled by low thrust, moderate efficiency EP, and thrust vectoring. "Moderate" efficiency from SRG connection to jet power (thrust $\times$ exhaust velocity^2 / 2) is 30–40%. This efficiency figure includes power conversion losses in the thruster PPU. The total thrust requirement to consider with this number is between 20–40 mN, divided among 2–4 thrusters to avoid plume impingement into the rings directly beneath the spacecraft, and to provide needed throttling between traverse and hover modes.
- Without technology development other than modest gains in small thruster efficiency to 55% (SRG to jet power), A ring missions can fit on Atlas V 551. AGA might allow using smaller Atlas V vehicles such as the 511 or 521.
- A level 1 science mission (final radius of 130,000 km) could be flown using any one of a wide number of technologies and architectures on an Atlas V 551. A level 2 science (120,000 km final radius) mission would require improved specific power for SRGs or higher-efficiency, low-power Hall thrusters .
- A 10–15 kW SEP stage or AGA could be used in many cases to move from the Delta IV Heavy launch vehicle to the smaller Atlas V family. This is also true for moving to smaller launch vehicles in the Atlas V family, but it is less likely to be cost-effective according to the launch vehicle costs stipulated in the Decadal Survey ground rules and recent studies on the costs of building large SEP stages. The Delta IV Heavy to Atlas V 551 jump is roughly $200M FY15 while each integer reduction of "x" in the Atlas V 5x1 family is only about $15M, which does not leave much funding for AGA development.

Note that all of the above conclusions require further analysis and optimization to confirm.

3. Technical Overview

Instrument Payload Description

In general, instruments that have already flown can be used to meet the observation requirements defined by the science team. However, the mass multiplier is quite large for this spacecraft, making low-mass payloads desirable. Additionally, the observational opportunities are rich, the scene is dynamic and, with the low altitude (implying close horizon), it is challenging indeed to predetermine and select scenes of interest via ground sequencing. Thus, this mission would benefit greatly from further technological developments in the areas of pattern recognition and automated sequencing. Similarly, navigation has challenges in hazard avoidance and altitude maintenance because of the variability and varying optical depth of the rings, and would also benefit from advances in pattern recognition and automated response.

Imaging

The science objectives of the SRO mission, as described in Section 1, could be achieved with imaging alone. The relevant measurements include particle velocities and size distribution, rotation rates, impact rates, and particle number density. The expected velocities are on the order of millimeters per second, the particle sizes range from one centimeter to meters, and the expected particle rotation period is on the order of 10 hours.

During measurement activities, the radial, tangential, and vertical velocity of the spacecraft would be near zero, relative to the average ring particle velocity. The low velocities suggest a framing camera would be preferable to a push-broom implementation. The nominal altitude above the rings is 2–3 km. The low altitude results in a close horizon; the maximum FOV (for reasonable angular distortion) is approximately 1 radian, subtending 3 km for a 3 km altitude. Measuring the velocity component perpendicular to the spacecraft would be difficult because of the low velocities and potentially irregular particle shapes. Taking stereo images would help, but would be challenging for this mission. The stereo baseline is typically 1/30 to 1/60 of the range (or 50–100 m). A 50 meter boom is possible, but impractical. The alternative could be a new activity of dithering the spacecraft while in the hover mode, which may prove to be challenging in terms of navigation and control.

The mean albedo of the ring is generally high and various. Lumme [6] reported one of the highest geometric albedos for Earth-based observations. The single scattering albedo also varies (particularly, it is believed, in the C ring). The value for single scattering albedo of 0.67 provided a good fit to HST data [7]. This high value suggests the surface of ring particles is young relative the ages of most icy surfaces in the solar system. It appears that the single scattering albedo has a large range of values for the C ring. The ring opacity varies from 0 (in gaps) to 1 (for dense rings). It is expected that individual particles, and shadowing on irregular particles, would be easily discernible and that it would be relatively easy to track particles with modest imaging rates.

The imaging goal is to achieve 1 cm resolution (~0.45 cm/pixel), though resolution of 10 cm (~4.5 cm/pixel) is acceptable. The spatial extent at this resolution should be as large as possible. The current typical flight framing cameras utilize a 1000 × 1000 pixel (small format) focal plane array (with spatial extent of 45 m for 10 cm resolution). Current commercial imaging chips are available in larger formats (4000 × 4000 pixel format is readily available) with spatial extents of 18 to 180 m, depending on the selected resolution. Similarly, a wide angle camera (WAC) (1 radian FOV) would cover 3 km for a 3 km altitude and, for large format, a 75 cm/pixel or ~1.65 m resolution instantaneous field of view (IFOV). There are many extant flight imagers with IFOVs in the range of 10–30 microradians. The throughput for the SRO imager does not have to be particularly high since the integration times can be long, on the order of seconds (though this interacts with the pointing stability requirements for the spacecraft). For most of the science investigations, panchromatic imaging is satisfactory. However, a filter set would be useful to aid in discriminating particle size. The current flight instruments used to size the narrow angle camera (NAC) and WAC are configured with 5 and 8 filters, respectively.

Tables 3-1 and 3-2 describe the NAC and WAC.

Table 3-1. Narrow Angle Camera

Item	Value	Units
Type of instrument: NAC	Hi-res imager	
Number of channels	5	filters
Size/dimensions (for each instrument)	30x20x20	centimeters
Instrument mass **without** contingency (CBE*)	8	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE+Reserve)	10.4	kg
Instrument average payload power **without** contingency	10	W
Instrument average payload power contingency	15	%
Instrument average payload power **with** contingency	11.5	W
Instrument average science data rate^ **without** contingency	16,000	kbps
Instrument average science data^ rate contingency	5	%
Instrument average science data^ rate **with** contingency	16,800	kbps
Instrument fields of view (if appropriate)	4.05	degrees
Pointing requirements (knowledge)	4e-4	degrees
Pointing requirements (control)	0.06	degrees
Pointing requirements (stability)	4e-3	deg/sec

*CBE = Current best estimate

^Instrument data rate defined as science data rate prior to on-board processing

Table 3-2. Wide Angle Camera

Item	Value	Units
Type of instrument: WAC	Wide imager	
Number of channels	8	filters
Size/dimensions (for each instrument)	5x5x5	centimeters
Instrument mass **without** contingency (CBE*)	0.5	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE+Reserve)	0.65	kg
Instrument average payload power **without** contingency	2	W
Instrument average payload power contingency	15	%
Instrument average payload power **with** contingency	2.3	W
Instrument average science data rate^ **without** contingency	12,000	kbps
Instrument average science data^ rate contingency	5	%
Instrument average science data^ rate **with** contingency	12,600	kbps
Instrument fields of view (if appropriate)	57.5	degrees
Pointing requirements (knowledge)	0.0058	degrees
Pointing requirements (control)	0.5	degrees
Pointing requirements (stability)	0.058	deg/sec

*CBE = Current best estimate

^Instrument data rate defined as science data rate prior to on-board processing

Data Generation

The nominal frame rate, during hovering mode, is one per minute, with expected mean particle motion during this interval of approximately 6 cm (and particle rotation of ~0.6 degrees.) A given particle should remain in the FOV for 750 minutes (somewhat more than one orbit) for a small format camera and four times that for a large format camera. For the 1 cm resolution, large format case, the particle would remain in the field of view for 300 minutes (~1/2 an orbit).

The raw data generation rate (16 bit word, panchromatic imaging) is 16 megabits/minute for the small format case and 256 megabits/minute for the large format case. The 16 bit data could likely be compressed lossless at 4:1. The nominal downlink rate is 30 kbps (up to 500 kbps) or 1.8 (up to 30 megabits/minute. For hover mode, 24x7 tracking is assumed. Thus, the small format case could be returned at about twice real time (real time [with margin]) and the large format case would require 35× (2.1×) real time (assuming a factor of 4 compression). The worst case (large format, low telemetry rate) would allow approximately 24 hours of observation for every 30 days of traverse. Observation duration would not be significantly restricted by telemetry for the other cases mentioned.

Other Instrumentation

Additional instrumentation would have significant scientific yield. As mentioned above, the mass multiplier for the hover spacecraft is rather large, so increases in the size of the payload affect significantly the ultimate inner radius achieved. (To be fair, the required wet mass increases steeply as the radius decreases—a greater factor than the dry mass of the hover spacecraft). Of the many cases described above, some successfully included a payload mass of 70 kg, which is sufficiently large to include many of the instruments mentioned below.

Science: Optical depth measurements (and particle size distribution estimates) would be improved by including images over a broader wavelength range. In particular, UV and IR images would measure the effects of smaller and larger particles, respectively. Instruments such as Alice (e.g., Juno or New Horizons) and Themis (MRO) provide reasonable examples of these classes of instrument.

Analysis of gas and particle composition (and particle velocity and size) is desirable. The gas environment could be measured with a mass spectrometer such as INMS (Cassini).There are no extant particle analysis instruments for these that are matched to the low particle velocities expected for this mission. The laser curtain approach of the Cassini Dust Analyser (CDA) would be suitable for measuring the velocities, but most proposed dust analysis instruments rely on high (>2 km/sec) impacts to ionize the particles for further analysis. Sample analysis instruments such as the Thermally Evolved Gas Analyser (TEGA; Phoenix) or the Sample Analysis Module (SAM; MSL), which includes a mass spectrometer and a tunable laser spectrometer, and laser ablation instruments (such as ChemCam on MSL) may not work well with the low particle densities expected at the spacecraft and have development issues with respect to sample collection. Future development is needed to support inclusion of a particle analysis instrument suited for the SRO mission.

There is also high value in understanding the electric field in the vicinity of the rings. A Langmuir probe or/and a plasma wave instrument (RPWS; Cassini) are likely candidates.

Engineering: Three engineering functions require additional instrumentation: maintaining a stable position with respect to a location on the rings, maintaining a fixed altitude, and performing hazard avoidance.

Maintaining a stable position is essentially an autonavigation-class activity. A reasonable implementation would require recognition of features in the ring, and maintenance position with respect to those features. This could be achieved with a single camera, though a stereo pair would improve the reliability of the required measurements.

The autonavigation approach could also be used to maintain altitude during observing periods. However, maintaining altitude during ring traverse is more complicated due to varying ring density and the limited horizon. Altitude could be measured with radar (and some yet-to-be-developed smoothing algorithm to determine the mean ring plane) or with a scanning LIDAR (light detection and ranging) with a similar

algorithm required (see Table 3-3). There are no suitable current flight-scanning LIDARs; however, there are excellent examples of (geophysical) field units used for mapping stratigraphy. The commercial units have a range up to 4 km and a resolution of centimeters and have reasonable mass and power parameters. Instrument development would be required to qualify such an instrument for flight.

Hazard avoidance would be needed to avoid wave structures and particles lying above the ring plane. Two approaches are reasonable, and both should be used. Imaging in the direction of traverse, parallel to the ring plane, should provide sufficient advance warning to terminate the traverse (or to increase the altitude). A medium angle camera would be suitable (~12 degrees FOV would resolve a 1.2 m object at 3 km). A scanning LIDAR would provide a similar function (at higher resolution) and provide ranging. If the LIDAR could also be operated in a LIDAR mode with a wide range gate, it would also provide estimates of particle number density distributions above the ring plane (expected to be small).

Table 3-4 provides payload mass and power for the NAC, WAC, and LIDAR.

Table 3-3. Scanning LIDAR

Item	Value	Units
Type of instrument: Scanning LIDAR (e.g., Riegl LPM-321; range to 6 km)	Wide imager	
Number of channels	1	Filters
Size/dimensions (for each instrument)	21x37x45	centimeters
Instrument mass **without** contingency (CBE*)	16	kg
Instrument mass contingency	50	%
Instrument mass **with** contingency (CBE+Reserve)	24	kg
Instrument average payload power **without** contingency	6**	W
Instrument average payload power contingency	30	%
Instrument average payload power **with** contingency	7.8**	W
Instrument average science data rate^ **without** contingency	1,000	kbps
Instrument average science data^ rate contingency	5	%
Instrument average science data^ rate **with** contingency	1,050	kbps
Instrument fields of view (if appropriate)	150x360	degrees
Pointing requirements (knowledge)	n/a	degrees
Pointing requirements (control)	n/a	degrees
Pointing requirements (stability)	n/a	deg/sec

*CBE = Current best estimate

^Instrument data rate defined as science data rate prior to on-board processing

Note: The LIDAR mass, power, and volume are for a large Earth-based system and have not been optimized for flight.

**Assumes a 10% duty cycle for scanning LIDAR.

Table 3-4. Payload Mass and Power

	Mass			Average Power		
	CBE (kg)	**% Cont.**	**MEV (kg)**	**CBE (W)**	**% Cont.**	**MEV (W)**
Instrument #1: NAC	8	30	10.4	10	15	11.5
Instrument #2: WAC (4 copies)	0.5	30	0.65	2	15	2.3
Instrument #3: Scanning LIDAR	16	50	24	6	30	7.8
Total Payload Mass	**24.5**	**–**	**35.0**	**–**	**–**	**–**

Note: Assumes a 10% duty cycle for scanning LIDAR.

Flight System

The hover spacecraft concept is generally based on the New Horizons spacecraft. The New Horizons analogy was applied to the avionics subsystems and the scaling of the structural mass. The telecommunications system was scaled up from New Horizons to support a more capable science return and, of course, the propulsion and power systems were deleted from the analogy. These were replaced with the power and propulsion systems that are candidates for future technology infusion.

Multiple flight elements were analyzed to estimate the mass, power, and launch requirements for the different science levels considered and the different technologies used to implement them. Rough mass and power analogies were considered to make initial estimates on what types of ring hover missions were feasible using different types of power and propulsion technologies. In addition to these analogies, simple parametric relationships were used to estimate structural and propellant tank masses for all flight elements. The delta-V and EP system propellant requirements were analyzed with consideration of different stages being jettisoned when they had performed all their intended maneuvers.

Technological requirements were examined through the use of "rubber" components (for sizing analysis) on the hover spacecraft. The EP thrusters were examined by the use of a simple jet power analysis. A single efficiency value covers the conversion of power from the outlet of the power source to jet power. Similarly, simply dividing required power by specific power yields the power source estimated mass. Both efficiency and specific power values were taken as technology metrics and were examined in this study.

Generic Hover Spacecraft Description

The hover spacecraft concept is designed to perform the science mission over the Saturnian rings. In addition to the onboard power and propulsion necessary to keep the spacecraft from descending into the rings, it would carry the full subsystem complement of a typical 3-axis-stabilized interplanetary spacecraft.

The hover spacecraft analogy to New Horizons is used primarily to establish mass and power estimates. Table 3-5 provides the flight system's subsystem masses. This analogy works well for subsystems such as the avionics that would have levels of functionality akin to New Horizons. However, the wide range of requirements on delta-V to be delivered by the electric propulsion system and scales of the "rubber" power system require a rubber structural mass. In this case, a simple factor upon the dry mass was employed because the power system mass varies as much or more than that of the EP or hopping propellant. This mass is more difficult to integrate within the primary load path than a centrally located propellant tank, and so should be thought of as more akin to distributed computer boards than a load of propellant. The value of this factor was chosen by considering the structural subsystem masses of multiple previously flown missions.

The EP version of the hover spacecraft would carry a vectored EP system that would have two major pointing modes. The first pointing mode is designed to create a resultant vector from the required hover and traverse thrust vectors. This would lead to a major reduction in required thrust (and thus power system mass) to achieve simultaneous radial traverse over the rings and hover. The second pointing mode is designed simply to hover over the rings with thrusters canted away from directly downward to avoid disturbing the ring particles. Since this was a technology study, a detailed examination of the thrust vectoring was not conducted. However, it should be noted that achieving the different pointing modes without incurring torques due to misalignment as the spacecraft's center of mass changes is non-trivial.

In addition to the electrical propulsion system used for the study, a small monopropellant chemical propulsion system was specified for the EP concept. The purpose of this system is to provide near-impulse burns to temporarily increase the spacecraft's altitude above the rings to avoid hazards. As discussed in Section 1, Saturn's rings are very thin except for in a few notable areas. If the EP system were sized to provide clearance over these obstacles, the power requirements would greatly increase stage mass and reduce the amount of radial traverse it could perform. Thus, the small chemical propulsion system is a more mass-effective solution to avoid these hazards.

Table 3-5. Mass Assumptions and Scaling Methods for Hover Spacecraft

Subsystem	Analogy / Sizing Method	Mass (kg)
ACS	New Horizons	19.6
CDH	New Horizons	17
Power electronics / distribution and batteries	New Horizons	28.4
Electric propulsion system	Scaled analogy from Team X studies for thruster, PPU, gimbal masses	Varies based on thrust, power
EP tanks	Fixed percentage	8% of propellant mass
Mono-propulsion system	Fixed percentage	12% of monopropellant mass
Structure	Fixed percentage	25% of dry mass
Mechanisms	New Horizons (allocated for telecom gimbaling)	23.1
Thermal	New Horizons	28.0
Telecommunications	Titan Saturn System Mission (high-performance system) study	64.0
Power source	Input variable (specific power)	See Key Trades section

An alternative to the EP system was considered using a bipropellant system. This is similar to the EP system analogy to New Horizons, except that a bipropellant system would be used to provide thrust to stay above Saturn's rings. Rather than thrust continuously, this system would "hop" with multiple bursts of thrust as described in a previous study [8]. This bipropellant system would still use either a reactor-based or radioisotope-based power system to power the spacecraft's subsystems.

The hover spacecraft was budgeted mass and power for a relatively high-performing telecommunications system. The system would have a 3 m antenna and a 50 W RF amplifier. This system would be capable of roughly 60–70 kbps from Saturn during downlink, which should be sufficient for downlinking a large number of images from the cameras in the science payload. However, as discussed later in the Concept of Operations and Mission Design section, this downlink rate is not sufficient for ambitious science objectives. While a detailed downlink and science operations scenario for the mission was not developed for this study, this telecommunications system was chosen as representative hardware for a mission that would ultimately be interested in high quality, moderately high-rate imaging to capture the dynamics of the ring particles.

The other subsystems for the spacecraft are assumed to be typical of a small exploration spacecraft bus. Attitude control and data handling subsystems are expected to demand relatively little mass and power. Structural and propellant tank masses are estimated as simple fractions of the dry and propellant masses, respectively.

As noted in the Key Trades section, many aspects of the spacecraft were left open to the choice of the designer to investigate the impacts on required technology metric values. This included the spacecraft average power, delta-V for hazard avoidance, and payload mass.

Cruise and SOI/Pumpdown Stages

No matter what level of science mission is addressed, the delivery from Earth to the rings of Saturn imposes major propulsion requirements. To fulfill them, at least one propulsion stage would need to be used with the hover spacecraft. In this section, a "cruise stage" refers to a stage that would provide post-launch propulsion from Earth to Saturn. A "SOI/pumpdown stage" refers to a supplemental stage that would deliver the hover spacecraft from Saturn approach to its rings.

A series of archetypical cruise stages were generated as concepts. The first is a generic chemical propulsion stage used to transfer the launch stack between Earth and Saturn, and to deliver the hover stage into its final orbit. Notionally, it consists of support structure, propellant tanks, and necessary thrusters and engines. It is built around a bi-propellant system with a specific impulse of 328 seconds, which is at the higher end of current hardware performance. Furthermore, 25% of the dry mass of the stage is allocated to structure. Another type of cruise stage is the SEP stage, which is patterned after a 15 kW stage used for other JPL concepts [2]. This stage is scaled linearly according to the required delivery mass.

Architectures using only chemical propulsion would use two chemical propulsion stages. This study did not address optimizing the delta-V allocation between those two stages. The optimal split might have a dedicated transfer (cruise) stage that handles maneuvers from launch to Saturn approach, and an SOI/pumpdown stage that handles all maneuvers from Saturn arrival (including SOI) to HOI. But the optimum might be a more even split of the total delta-V budget, so one stage would handle, for instance, launch through the end of the leveraged pumpdown, and the other would handle the large maneuvers from the end of pumpdown to HOI. This trade study would be an important part of future studies of SRO mission concepts.

Architectures with the SEP concept would use both the SEP stage and a chemical propulsion stage. In this case, the second propulsion stage would be used to perform the SOI and moon leveraging tour, as well as the final large maneuver to reach HOI.

The Titan AGA architectures would utilize an aeroshell to dissipate kinetic energy via drag in Titan's atmosphere. The aeroshell would be shed, leaving the spacecraft in Saturn orbit. A chemical delivery stage would then perform the leveraging tour of the moons and the final maneuvers leading to HOI.

Finally, a cruise stage architecture using both SEP and AGA to achieve Saturn orbit was considered. This architecture would utilize SEP to reduce the C3 energy requirements of the launch vehicle and increase useful mass at Saturn. The AGA would reduce the required delta-V to reach HOI on the chemical stage.

Both large SEP stages and AGA are technologies that would require further maturation before being used on a flight project. Their contributions to the ability to perform more aggressive Saturn ring missions are discussed in Section 2, Key Trades.

None of the above masses in Tables 3-6 and 3-7, nor the masses for the hover spacecraft, have contingency applied. Instead, it was required that the margin between the estimated launch stack mass and the launch vehicle delivery capability be 30%.

Table 3-6. Mass Assumptions and Scaling Methods for Chemical Cruise Stages

Subsystem	Analogy / Sizing Method	Mass (kg)
Power Electronics / Distribution and Batteries	Power electronics allocation, small battery	20
Propulsion Hardware (thrusters, valves)	Estimate between New Horizons and MRO	40
Chemical propellant tanks	Fixed percentage, matched to analogy based on delta-V capability	8% of propellant mass
Structure	Fixed percentage, matched to analogy based on delta-V capability	25% of dry mass
Mechanisms	New Horizons-based allocation	20
Thermal	New Horizons	30
Cabling	New Horizons-based allocation	25

Table 3-7. Mass assumptions and scaling methods for solar electric propulsion stage @ 15 kW power; linearly scaled with launch stack mass from 6,300 kg reference point

Subsystem	Analogy / Sizing Method	Mass (kg)
Power Electronics	TSSM [2]	10
Propulsion Hardware (thrusters, valves, PPU)	TSSM	86
Solar Array	TSSM	53
Xenon tanks	Fixed percentage	8% of propellant mass
Structure	Fixed percentage	25% of dry mass
Mechanisms	TSSM	36
Thermal	TSSM	38
Cabling	TSSM	34

Power for individual stages was not analyzed. Rather, the hover spacecraft bus power was treated as an input parameter for study, which was eventually set at 350 W. This was taken by analogy to small spacecraft powers from previous studies with ample margin set aside for regular data downlinks with a highly capable telecommunications system. Cruise stage power was not considered an important factor due to the large quantities of available power in the hover spacecraft. The solar electric propulsion stage was linearly scaled up or down from a baseline design capable of generating 15 kW for its own use.

Technology Description

Power Technology

Two technology architecture approaches were considered for SRO spacecraft bus and hover/transfer propulsion power, ASRGs and small fission power systems (FPSs). ASRG and small fission power systems are the two power systems addressed.

Advanced Stirling Radioisotope Generators

RPSs have been used for many NASA missions, including Voyager, Cassini, and New Horizons. This technology converts the heat output by a radioisotope, plutonium 238 (^{238}Pu), into electrical energy. This conversion has been achieved through the use of radioisotope thermoelectric generators (RTGs) and in the next generation multimission radioisotope thermoelectric generator (MMRTG) that will be used in the Mars Science Laboratory (MSL). One candidate for future NASA missions is the ASRG, shown in Figure 3-1, currently under development by Lockheed Martin, the Department of Energy (DOE) and NASA. This ASRG would provide for a significantly higher thermal to electric power conversion efficiency. Current estimates are that ASRGs would produce 146 W at beginning of life (BOL) with a mass of 24 kg [9]. NASA's Science Mission Directorate (SMD) is moving toward the use of ASRGs on the 2016 Discovery 12 mission. ASRG characteristics are defined in [10] as a ground rule to Decadal Survey studies.

Initial power requirements for the SRO range from 1 to 3 kWe, which would result in a large number (7–21) of ASRG units with the associated issues in spacecraft integration and potentially unit production capacity. While the development of the 146 W ASRG is most mature, higher power versions are possible and studies have shown that a 550 W class SRG would have potentially superior specific power and may ease integration requirements into a spacecraft of this class [11]. If a 550 W class generator were developed, only two to six SRGs would be required, simplifying spacecraft design while improving power system performance. Advances in the current ASRG providing higher specific power may also be available for SRO concepts with power requirements at the low end of the above range. Table 3-8 provides overall characteristics for the current ASRG, the 550 W SRG, and an advanced, optimized version of the current ASRG.

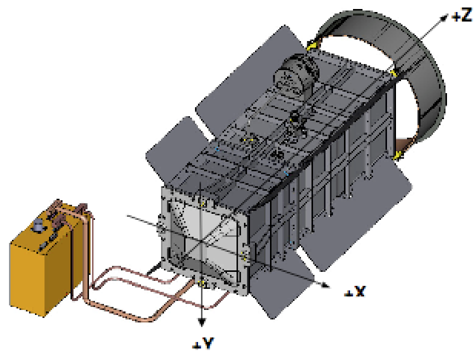

Figure 3-1. Advanced Stirling Radioisotope Generator

Table 3-8. Stirling Radioisotope Generator Options and Characteristics

	Current ASRG (non-optimized)	550 W SRG (estimated)	Optimized ASRG (estimated)
System Life (years)	14+	14+	14+
BOM Power (watts)	145	550	160
Power at 5 years	141	531	153
Power at 10 years	133	512	146
Power at 14 years	126	499	142
# GPHS Modules	2	6	2
Mass (kg)	24	54	22
BOM Efficiency (%)	29	~35	32
BOM Specific Power (w/kg)	6.0	10.2	7.3
EOM Specific Power (w/kg)	5.3	9.2	6.45
Dimensions			
Diameter (cm)	34	53	34
Length (cm)	77	85	77

The 550 W SRG would be an engineering modification of the advanced Stirling converter but would require a new design. The same Mar-M 247 heater head material would be used as well as the same tools and Stirling configuration that is used in the ASRG convertors. Moving to higher power levels is challenging due to the integration of the low heat flux general purpose heat source (GPHS) modules and the high heat flux requirements of the Stirling hot and cold end. Because of the geometry constraints and these heat flux mismatches there are practical limits as to how many GPHS modules can be conductively connected to a Stirling converter. As the number of GPHS modules grow, the mass of the heat connection flange and its temperature losses increase, at some point making this type of integration impractical. A design approach was selected that keeps the conductively coupled heater head rather than moving to either heat pipes or pumped loops. Figure 3-2 shows this conceptual design. Studies have shown that specific power peaks at this 550 W power level; higher-power SRGs, at reduced specific power, are feasible if mission requirements warrant.

Development of the 550 W SRG design would leverage many of the principles to be demonstrated by the first flight of the 146 W ASRG. Although the 550 W SRG would be larger and contain some different materials, it would also leverage the controller design developed by the 146 W ASRG. An in-depth investigation of the 550 W SRG should be carried out upon completion of the 146 W ASRG to thoroughly characterize the development project. Development of the 550 W SRG is projected to require approximately seven years. There is no current work being carried out on the 550 W ASRG by NASA. Development of the optimized 160 W ASRG, shown in Table 3-8, would require approximately five years.

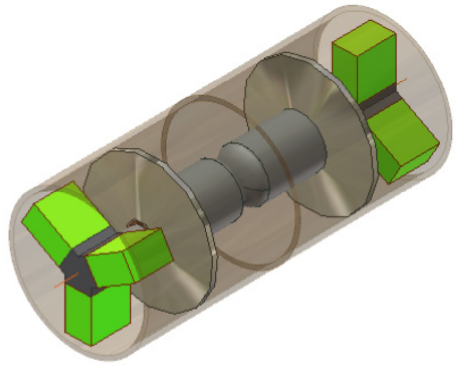

Figure 3-2. Conceptual Design for 550 W SRG, Showing Six GPHS Modules

Two risks were identified relating to the potential use of Stirling RPSs for the SRO. The first risk identified is the possibility of multiple Stirling converters mounted on the spacecraft to generate a beat frequency related to the Stirling converters' interactions with each other. Each Stirling RPS, while operated in such a way to minimize vibration, is not designed to be synchronized with other Stirling RPSs. One possible way to reduce or eliminate this is to design the controllers to allow for synchronization of all of the Stirling RPSs on the spacecraft. The controller for the ASRG is not designed to allow this synchronization.

The second risk is ^{238}Pu supply. Currently, there is a limited supply of ^{238}Pu available for NASA missions. The DOE is currently considering restarting production. At the beginning of the 2012 PSDS mission studies, the anticipated amount available to NASA was 5 kg/year, though since then the DOE has discussed amounts as small as 1.5 kg/year [12].

Small Fission Power System

The SRO study team coordinated with the small FPS study team to consider the application of that technology to the mission concept [13]. The main objectives of that study were to evaluate the feasibility of a small FPS for NASA science missions and to provide information to the Decadal Survey Giant Planets Panel that would guide their recommendations to NASA concerning an investment in a small FPS. The primary motivation was to identify a power system option for the larger flagship science missions whose power requirements may exceed what is practical with the current suite of 100 W class RPSs.

The team developed a set of notional requirements, which were accepted by the Giant Planets Panel and which included 1 kW electrical output, 15-year design life, and 2020 launch availability. After completing a short round of concept screening studies, the team selected a single concept for further study and analysis. The selected concept, illustrated in Figure 3-3, is a uranium-molybdenum fueled, heat pipe cooled reactor with distributed thermoelectric power converters coupled directly to aluminum radiator fins. The team generated a preliminary configuration, mass summary, development schedule, and cost estimate. The system mass is 772 kg (including margin) and the rough order-of-magnitude cost for the 10-year flight system development program is $690M. The concept is described in full in the reference study report.

Application of this technology to the SRO mission was not optimal. The SRO spacecraft to be inserted into the ring science orbit would be low mass and would need a low mass, efficient power and propulsion system to effectively maintain the hover orbit. The small FPS concept specific power of 1.7 W/kg would prevent the mission solution from closing for available launch system concepts. Minimal specific power values for a feasible hover spacecraft are identified in Section 2, Key Trades. The small FPS concept remains an option for future evaluation under the following considerations: difficulty in obtaining ^{238}Pu thereby constraining SRG capability, and/or development of the small FPS as a multimission capability, thereby facilitating possible implementation on a significantly different SRO mission concept.

Other design points for the reactor were also considered in support of finding an appropriate size for the SRO concept. The power levels and masses of the reactor are provided in Table 3-9.

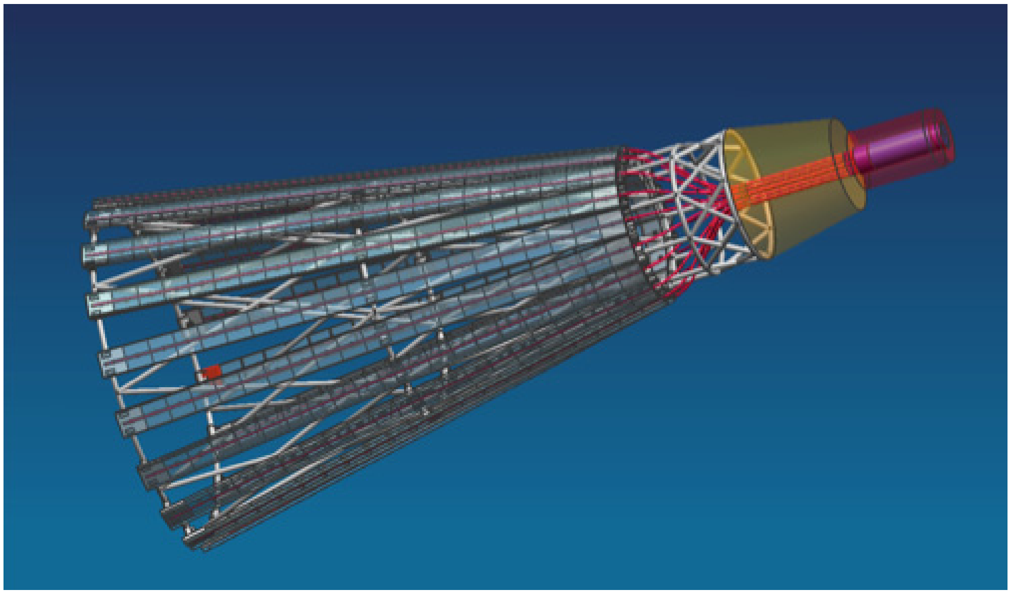

Figure 3-3. Conceptual Design for Small Fission Power System (1 kW)

Table 3-9. Small Fission Power System Sizing

	Thermoelectric				Stirling		
Power, kWe	0.5	1.0	1.5	3.0	3.0	5.0	10.0
Reactor, kWt	6.5	13.0	19.5	39.0	13.0	21.7	43.3
Radiator area, m^2	2.5	5.0	7.5	15.0	8.0	13.3	26.7
Radiator length, m	1.3	2.5	3.8	7.5	4.0	6.7	13.3
Separation distance, m	10.0	10.0	10.0	10.0	10.0	10.0	10.0
Mass, kg							
Reactor	178	184	189	239	184	184	220
Shield	195	272	330	463	272	348	487
Balance of Plant	84	149	214	410	370	604	1,190
Total, kg	457	605	733	1,112	826	1,136	1,897
Specific Power W/kg	1.1	1.7	2.0	2.7	3.6	4.4	5.3
Including Margin, kg	580	772	940	1,428	1,059	1,462	2,448
Specific Power W/kg	0.9	1.3	1.6	2.1	2.8	3.4	4.1

Propulsion Technology

Potential propulsion technologies for the SRO mission concept are of two broad types: those applicable to the hover and traverse functions of the hover spacecraft, and those applicable to delivery of the hover spacecraft to HOI. Those applicable to hover and traverse are discussed first, followed by the delivery technologies.

The total power for a given science level is explored in Section 2, Key Trades. SRO preliminary mission analysis indicated that to perform the traverse and hover maneuvers for the first science level would require an EP system that could operate at power levels between 300 to 600 W, has a thrust-to-power (T/P) ratio ≥100 mN/kW with a thrust efficiency ≥40%, a total ΔV of 1.5 to 3.5 km/s, and a PPU efficiency ≥90%. These values define a technology development for new low-thrust units. Having a thruster with higher T/P capability and/or higher thruster efficiency would result in greater mission benefits. Table 3-10 indicates that for a given power level, improving thruster efficiency would result in higher specific impulse and a lower propellant load for the SRO mission.

Table 3-10. EP Thruster Specific Impulse for Various Thruster Efficiencies

Thrust/Power = 100 mN/kW	
Efficiency	**Specific Impulse, s**
0.30	612
0.35	714
0.40	815
0.45	917
0.50	1,019
0.55	1,121
0.60	1,223
0.65	1,325
0.70	1,427
0.75	1,529

A number of EP thruster technologies were evaluated to determine their potential implementation and suitability for the SRO mission. Thruster technologies evaluated included arcjet, ion, Hall, and colloid thrusters. Survey of potential low power arcjets indicated that although the T/P of the devices meets/exceeds the SRO mission concept requirements, thruster efficiency does not meet the SRO mission concept requirements. Evaluation of ion engine (gridded) technologies indicated that they are not capable of achieving the desired T/P at the operating power range of 300–600 W. Evaluation of state-of-the-art Hall thruster technology indicated that achieving the desired EP system performance has been demonstrated with a laboratory thruster and would be achievable with additional investment in the technology with the caveat that thruster efficiency would ultimately not exceed 50–60%. The final electrostatic EP technology that was evaluated was colloid thrusters. Colloid thrusters have the potential to meet the T/P requirements with thrust efficiencies as high as 80%, but would require significant development investment and time. As such, a Hall thruster system was baselined for the SRO mission concept because of its flight heritage and lower risk technology development plan. The following sections provide a description of the baselined Hall thruster propulsion system and what technologies need to be incorporated to achieve the maximum EP system performance. A discussion of colloid thrusters is also provided to highlight its mission benefits.

Hall Thruster Development

NASA, the Department of Defense (DoD), and industry have extensive experience in developing and flying Hall thruster propulsion systems. Aerojet, Busek, and Loral have flown a number of Hall thruster systems with varying power levels but with T/P ≤100 mN/kW and xenon throughput capability that is less than 300 kg of xenon.

DoD and industry have been pursuing the development of Hall thruster designs with high specific impulse and high T/P capability. DoD and its industrial partners have demonstrated a laboratory Hall thruster with T/P levels ~100 mN/kW with thrust efficiencies that meet the SRO mission concept requirements. In addition, a NASA Glenn Research Center/Aerojet team has been developing a high-voltage Hall Accelerator (HiVHAc) flight-like thruster that incorporates a life extending mechanism to enable xenon throughputs in excess of 300 kg (needed to achieve total mission ΔV). As such, the development of a new Hall thruster that meets the SRO mission concept requirements would involve integration of tested thruster configurations and innovations that have already been demonstrated by different Hall thruster development programs. In addition to the above thruster developments, NASA, DoD, and industry have been pursuing the development of low-mass high efficiency PPU and propellant feed systems for incorporation in near-term DoD and NASA science missions. Figure 3-4 shows a photograph of a flight-like 600 W Hall thruster.

Figure 3-4. Photograph of Flight-Like 600 W Hall Thruster

Further investment in specific technologies could result in an even more efficient Hall thruster system as indicated in Table 3-11. Phase 0 represents advances currently in development within the In-Space Propulsion Technology project. Phase 1 and 2 technology advances include:

- Development of more efficient (both in power and propellant consumption), lower mass, and smaller size hollow cathode assemblies; 3–5 year development effort
- Development of propellantless cathodes; 10–15 year development effort
- Development of magnetic alloys with enhanced magnetic properties; 3–5 year development effort
- Development of electromagnets with lower ohmic loses; 3–10 year development effort
- Development of permanent magnets with field strength superior to SOA; 10–15 year development effort
- Development of new PPU topologies with efficiency ≥ 98%; 5–10 year development effort

Table 3-11. Hall Thruster Development Options

	η	**T/P**
Present (SOA)	45–48%	80–90
Phase 0 1–2 years	50%	100
Phase I 3–5 years	55%	>100
Phase II 5–10 years	60%	>100

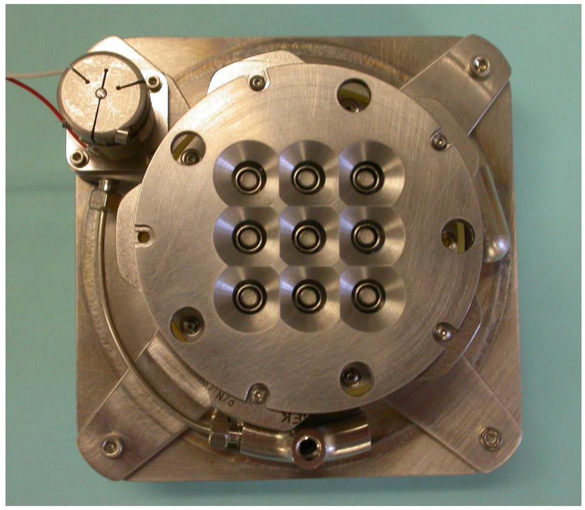

Figure 3-5. Photograph of a Laboratory 10 W, 1 mN Colloid Thruster

Colloid Thruster Development

Colloid thrusters are an attractive option for use on a future SRO mission due to their variable specific impulse and variable thrust capability with T/P levels that exceed 100 mN/kW and a thrust efficiency ~80%. Current state of the art (SOA) colloid technology has demonstrated a ~10 W thruster system with a T/P ~100 mN/kW. Figure 3-5 shows a photograph of the thruster.

Scaling up the power level to ~200 W is being pursued by industry. A 200 W colloid thruster system would result in an EP system that meets and exceeds the SRO mission concept requirements and would result in a reduced power requirement to meet the SRO mission concept requirements. To be able to demonstrate a 200 W colloid system requires investment in the following technologies:

- Demonstration of scaling from circular to annular emitter configuration; 2–3 year development effort
- Material development for life demonstration; 3–5 year development effort
- High efficiency 150 W, 10 kV power supply; 3–5 year development effort
- Neutralizer scaling to 15 mA; 3–10 year development effort
- Propellant feed system scaling; 3–5 year development effort

Propulsion technologies associated with delivering the hover spacecraft to HOI would apply to the two mission phases preceding HOI, the launch/transfer phase and the SOI/pumpdown phase. Hover orbit phase propulsion technologies, specifically REP and NEP, were considered, but at power levels appropriate for the hover orbit; neither contributed to significant mass savings or flight time reduction when used for the other phases. The principal technology considered for the launch/transfer phase is SEP, but future studies should also examine the benefits of alternate launch modes, by which adjusted timing and trajectories of upper stage burns can yield improved performance. For the SOI/pumpdown phase, the principal technologies considered are high-I_{SP} chemical propulsion and aero-assist.

SEP has been in use for many years on geostationary satellites and is already in use for planetary science missions, including the New Millennium Program's DS-1 mission to comet Borrelly, ESA's SMART-1 mission to Earth's moon, and the Discovery Program's Dawn mission, currently en route to orbit asteroids Vesta and Ceres. With technology advancement, SEP promises benefits to a larger range of missions. Science missions to date using SEP have operated only one EP engine at a time. Minor technology development would enable SEP systems that operate multiple EP engines simultaneously, a cross-cutting capability that would make them useful for a wide range of missions needing larger spacecraft. Such applications would also benefit from engines with higher propellant throughput capacity, such as the NEXT engine currently under development, and from solar electric power systems with higher

specific power, such as the 15 kWe UltraFlex arrays being developed for the Orion spacecraft. The TSSM study [2] considered such a system and found it to be an effective means of increasing the mass delivered to Saturn in years when JGAs are not available.

Although development of a SEP stage would benefit many missions, use of SEP does not *require* a particular SEP stage. SEP stages could be engineered to provide flexibility in the number of engines, electric power levels, and propellant load, in a single package with a simple interface to the spacecraft, with the capability to be jettisoned if that is advantageous. For some missions, the greatest benefit would be derived from having a SEP "kit" that would be integrated with the host spacecraft. The kit could be a set of box-level components designed to operate together properly, so the spacecraft engineers would need only to design the size of the system (i.e., number of components) and the spacecraft's accommodation of those components, and not design the details of the entire SEP system.

Upon Saturn approach, high-I_{SP} chemical propulsion could decrease the mass fraction devoted to the SOI/pumpdown stage, either to make more mass available to the hover stage or to reduce the size and cost of the launch vehicle needed. Technologies currently under consideration for high-I_{SP} in-space chemical propulsion include exotic non-cryogenic propellants (that generally require exotic tank and plumbing materials), such as tri-propellant systems, and space storage of cryogenic propellants. All these technologies involve increasing the inert mass fraction of the propulsion system, an almost unavoidable consequence. But despite that initial penalty, increasing I_{SP} from the current state-of-the-art ~330 s to 400 s could save more than a metric ton of SOI/pumpdown stage mass for a 1-ton (wet) hover spacecraft and all-chemical SOI and pumpdown. Currently, the TRLs for exotic propellant systems are quite low, about TRL 2, mostly due to issues concerning stability of propellant storage and delivery, *i.e.* operating without exploding. An entry-level cryogenic system, using LOX as the only cryogenic part of the propellant mix, is at TRL 3. Laboratory demonstration systems have successfully used hydrazine as the fuel and actively cooled LOX as the oxidizer, with zero boil-off loss. This mixture falls somewhat short of the 400 s I_{SP} mentioned above, with a capability of about 380 s and concomitant reduction in mass savings. Increasing I_{SP} to 400s or larger would require either an exotic fuel (at lower TRL) or LH2. To date, rocket propulsion researchers have not identified a practical means of achieving zero boil-off with LH2.

Aero-assist methods are alternatives to the high-I_{SP} chemical propulsion methods. This study considered two different types of aero-assists: aerocapture into Saturn orbit using Saturn's atmosphere, and AGA into Saturn orbit using Titan's atmosphere.

Aerocapture in Saturn's atmosphere is a very demanding task. Use of this method to achieve HOI was studied previously [8, 14] as an alternative to an all-NEP implementation of an SRO mission. Because this occurs far down in Saturn's gravity well, the velocities and delta-Vs involved are extremely high. Relative to Saturn's rapidly rotating atmosphere, the vehicle's entry velocity would be ~26 km/s, and the necessary delta-V (for an SRO mission) would be ~7 km/s. Due to the extreme aero-heating environment of this entry, the aeroshell needed for such a maneuver constitutes 40–50% of the Saturn approach mass [15], and must be a high-L/D design (L/D ~0.7 to 0.8) that currently has not progressed beyond basic-research TRLs. After shedding the aeroshell, there would be still be propulsive maneuvers of significant total delta-V to be done. Immediately after releasing the aeroshell, there would be a maneuver to clean up the residuals from the aerocapture, with a delta-V potentially as large as 500 m/s. Approximately 3.5 hours after the aerocapture maneuver, there would be another maneuver to circularize the orbit (and avoid impacting the rings) at the post-aerocapture apoapse, with a delta-V of ~3 km/s. This total delta-V would be very similar to the total delta-V for an all-chemical SOI and pumpdown using leveraged propulsive pumpdown that requires no additional large aeroshell. Use of the leveraged propulsion technique is new since [14], and essentially makes that Saturn aerocapture approach obsolete. Other variants using Saturn aerocapture were examined in this study, but the post-aerocapture propulsive requirements for those were at least as demanding as for the original approach, most even more so. Thus, all the Saturn aerocapture approaches considered required more, not less, mass delivered to the Saturn system than the much more technologically mature all-chemical approach.

The other aero-assist method, AGA into Saturn's orbit using Titan's atmosphere, fares much better. Because Titan is much more distant in Saturn's gravity well, the relative velocities are much lower, with typical Titan atmospheric entry speeds from Saturn approach of 9–12 km/s instead of the 26 km/s into Saturn's atmosphere. Titan's small size relative to Saturn gives it a much lower gravitational acceleration,

so Titan's atmospheric scale height is larger than Saturn's at the altitudes of interest for deep (i.e., not merely aerobraking) aero-assists. These differences make possible a much higher-TRL aeroshell, with L/D ~0.2 to 0.25, a range for which there are designs at TRL 7 and higher, that uses high-TRL thermal protection system materials. The Titan Explorer study of 2007 [16] studied aerocapture from Saturn approach directly into Titan orbit and found that at a delta-V of ~4 km/s, this could be accomplished with an aeroshell mass fraction of ~27%. Titan AGA into Saturn orbit would be less demanding, with a similar entry speed but a delta-V of only ~1 km/s. This study used an aeroshell mass fraction of 0.25, but there is potential for that to be smaller. Nonetheless, using that mass fraction, performing SOI and pumpdown with a Titan AGA and then chemical pumpdown at an I_{SP} of 330 s would save ~1.5 tons of mass delivered to Saturn approach (for a 1-ton hover stage), as compared to the all-chemical SOI/pumpdown at the same I_{SP}. This is more mass savings than an all-chemical SOI/pumpdown at an I_{SP} of 400 s, and roughly equivalent to the mass savings at an I_{SP} of 440 s.

Operations Technologies

The SRO science mission phase, the "hover orbit," involves a new mode of operation—station keeping with a nebulous, at times poorly spatially-defined object, Saturn's rings. There are times that this could be done "open-loop." By controlling the magnitude and direction of the EP thrust vector, the spacecraft would establish where the physics of the system says it should go. At other times, such as when the rings themselves or the spacecraft are being perturbed by forces such as those arising from resonances with moons, a "closed-loop" approach might be needed. At the simplest level, the closed-loop task would require measuring the distance to the ring particles, inferring the location of the mean ring plane, and planning and executing the changes in hover thrusting to place the spacecraft where it is supposed to be.

Such simplicity is complicated by the non-rigid and highly variable character of the rings. In some locations, like the densest parts of the B ring, the view of the rings from 2–3 km out of the mean plane looks something like a rough but solid surface; an electromagnetic probe signal, such as a LIDAR beam, gets a return essentially from everywhere. But in many locations, there is sufficient space between particles that the returns are intermittent. In some locations, the particle density is so low that a fixed-beam LIDAR might go long periods without getting any return at all. This argues for a two-dimensional probe rather than a point probe of the rings, possibly a scanning LIDAR. Another technological need concerns the algorithms by which a spacecraft would infer from the instrument data the location of the ring plane, and determine the spacecraft's response to that knowledge. Those algorithms would have to be developed and thoroughly tested before the use of this system would pass a review board.

Potentially, there are engineering and science functions this system might need to perform beyond the range-determining function, and they are a bit more complex. One such engineering function would be to detect the lateral movement of the spacecraft with respect to the aggregate motion of the ring particles beneath it, so the spacecraft can truly establish a "motionless" (with respect to those ring particles) hover and observe those particles for extended periods. Merely interrupting EP traverse thrusting does not establish this motionless hover; during EP traverse the spacecraft has an inward velocity component in the range of 0.1 to nearly 2 m/s, so at traverse thrust termination the osculating orbit is slightly eccentric. Over one orbit period this would have the spacecraft oscillate inward and outward one to several kilometers, so the NAC would not see one scene for sufficient time during critical particle-to-particle collision observations. Detecting the lateral motion with respect to the particles would be the first step toward having the spacecraft circularize the orbit via the EP thrusters, and actively control the motionless hover. Another engineering function might be detecting hazards, such as significant, coherent excursions of the ring particles from the mean ring plane ("warping" of the rings) that should prompt the spacecraft to take evasive action. A science function would be to determine the out-of-the-plane components of ring particle velocities, a task that might be done with stereo imaging, but with a significant interocular distance and significant increase in data rates. Algorithms for handling all these functions would have to be developed. If established, the need to perform these functions might also complicate the design of the instrument used for range determination.

Another science function, one that might make use of imaging or LIDAR data or both, would be autonomous detection and response to unexpected targets of opportunity during EP traverse. Such targets would be phenomena, geometries, albedo variance, or other things of great scientific interest

whose appearance in the instruments' field of view could not be anticipated by the science or operations teams. It is clear that if the ground operations team must be in the loop, detection of such targets and the decision to return to them happen far too slowly for a return to the location of the target (see Concept of Operations section). By the time commands to return to a given location arrive at the spacecraft from the ground, the spacecraft resources required to return (propellant and mission time) are excessive. Onboard autonomous routines to sift through data being acquired as the spacecraft traverses (data far too voluminous to be downlinked) could detect the presence of patterns pre-defined in the software that would trigger a canned sequence to "stop and look." Within seconds of the trigger command, this sequence would stop the traverse and circularize to a full hover over the location in the rings that caused the trigger, and at the next downlink opportunity would notify the operations team that something very interesting has been encountered.

Concept of Operations and Mission Design

Concept of Operations

The three high-level SRO mission phases, launch/transfer, SOI/pumpdown, and hover orbit, all have multiple options for their implementations, but the multiple options converge at the transitions from one phase to the next. This allows discussing operations for all mission options, one phase at a time. The transitions are Saturn approach, where the launch/transfer phase links to the SOI/pumpdown phase, and HOI, where the SOI/pumpdown phase links to the hover orbit phase. Different Earth-to-Saturn transfer trajectories introduce variability into the Saturn approach, since different transfers would differ in the two most important parameters, V-infinity of approach and declination of the approach asymptote. Despite these variations, the Saturn approach would be steered to feed smoothly into SOI, so the variability of approach circumstances is unimportant at the architectural level. There is much less variability in HOI circumstances, essentially only the relative longitude of the spacecraft with respect to some other feature of the Saturn system, such as the location of a nearby ring moon, or a particular solar phase angle. The specification of precise trajectories is a detail beyond the scope of this study. The highlights of different mission options are discussed in this section in the order in which they would be executed for a mission.

In all cases, the flight mission would begin with launch, most likely from one of the launch complexes at Cape Canaveral, Florida. The study identified potential mission options that might launch on vehicles ranging from an Atlas V to a Delta IV Heavy. Since this mission would rely on nuclear power sources, the launch facility would need to support nuclear launch operations. For an RPS-powered mission, these operations should break no new ground, but for a fission-powered mission, new procedures and possibly even new facilities could be involved. It is highly unlikely SRO would be solely responsible for these new procedures or facilities, since the development of fission power systems would be under a program involving multiple flight missions.

The type of Earth-to-Saturn transfer would determine the launch C3. The study identified two main variants of transfer trajectories. One would use inner solar system gravity assists to reach Jupiter and then a JGA to reach Saturn, with all propulsion from a chemical system. There is a type similar to the first but that does not use the JGA because the Jupiter-Saturn alignment at the mission epoch does not permit it. The long transfer duration of most of these make them unattractive options, but a few might be considered under some circumstances. The other main type would be a SEP-assisted trajectory that would use one or more inner solar system gravity assists, and might or might not use a JGA. Non-SEP trajectories would launch to a C3 in the 9–50 km^2/s^2 range. At the low end are trajectories that would first go to Venus. In the middle are 2-year Delta-VEGA trajectories that would do an Earth gravity assist ~2 years after launch. At the high end are 3-year Delta-VEGA trajectories. Since lower C3s yield larger launch capability, the more demanding missions tend toward these trajectories. SEP trajectories could launch to C3s only slightly larger than zero.

With one exception, operations during the transfer phase should not deviate significantly from processes already understood. Operations support for the non-SEP trajectories is fairly standard, with plenty of predecessor missions. Procedures for such methods as trajectory correction maneuvers (TCMs) and aim-point biasing for Earth flybys are well understood. Operations procedures for a SEP-assisted transfer with a large SEP stage simultaneously running multiple thrusters is new, but the TSSM concept study

examined this and found no significant problems [2]. The exception would be a fission-powered mission that starts its reactor at some time before Saturn approach. No NASA interplanetary mission has ever operated a nuclear fission reactor power system in space. The technology and operations processes for doing so would need infusion into the NASA community. If the reactor is started soon after launch, there might be planetary protection and thus operations ramifications for subsequent Earth flybys, as well as for Titan and Enceladus flybys and other operations within the Saturn system. In all cases, the transfer phase would deliver the stack to Saturn, ready for insertion into Saturn orbit. Final delivery to Saturn would involve several months of approach operations, including navigation activities and TCMs to steer to the proper approach state (V-infinity, declination, and B-plane aim point).

The SOI/pumpdown phase begins with SOI, after the handoff from the transfer phase. The study identified two preferred methods for SOI: chemical propulsive insertion or Titan AGA. Other methods were examined but deemed less effective than these two. The first is the standard approach, used by NASA for a host of past planetary orbit insertions. Operations for a chemical SOI are well understood, with some variability involved depending on the Saturn approach state derived from the transfer trajectory. Typically for a giant planet orbit insertion, the first orbit after SOI is a long-period orbit (~100–200 days) to save propellant; insertion into shorter-period orbits uses significantly more propellant. At the apoapse of that orbit, a propulsive periapse raise maneuver (PRM) would raise the orbit's periapse to avoid ring hazards. For SRO, that PRM would also target a first Titan gravity assist flyby to begin an "initial pumpdown" during which the spacecraft's orbital energy would be reduced by approximately three Titan gravity assists. The objective of the initial pumpdown is to place the spacecraft in an orbit with apoapse near Titan and periapse at Rhea to start the "interior pumpdown" ("interior" to Titan's orbit). Operations for the initial pumpdown involve navigation and TCMs to steer the Titan gravity assist flybys, and standard spacecraft health monitoring.

A Titan AGA could skip the initial pumpdown and deliver the hover spacecraft and pumpdown stage directly to the start of the interior pumpdown. Operationally, the aero-assist part of the AGA is quite simple; events happen far too quickly to have the ground in the loop, so the team can only watch as the data come down to Earth, some two hours or more after it is all over. Before the AGA, there would be the standard system checkouts, navigation activities, and TCMs. After the AGA, there would be more system checkouts and navigation activities, and a potentially large propulsive maneuver to cancel velocity residuals from the AGA and initiate the interior pumpdown.

Whether SOI is done propulsively and followed by an initial pumpdown, or is done via an AGA, the interior pumpdown that follows would be the same in character, though details might differ. Duration and total delta-Vs would be very nearly the same. The time-consuming and operationally intense part would begin with the first of a rapid series of flybys of Rhea, Dione, Tethys and Enceladus, with leveraged propulsive maneuvers between. If flown such that the pumpdown duration is minimized, the periods between flybys could become very short (a few days). This would heavily tax current operations processes for handling all the navigation and TCM tasks. Pumpdown designs used in this study limit the brevity of time between flybys to no less than three days, judged to be possible with enhancements to processes that the Cassini Project currently uses. At the end of this series of flybys, some 3 to 3.5 years after chemical SOI (~6 months less for AGA), the spacecraft's orbit would have apoapse near Enceladus' orbit, and periapse at a safe distance outside Saturn's F ring at a radial distance of ~145,000 km. It is possible that science observations might be made during these flybys. If so, generation of science observation sequences and downlink of science data would be added to the operations workload.

At the end of the interior pumpdown, no more useful gravity assists would be possible, so all subsequent delta-V leading to HOI would be propulsive. Transfer from the final pumpdown orbit to HOI could be done with one or two steps. The one-step approach would have a propulsive maneuver at apoapse move periapse across the F ring to a safe position between the F and A rings, using a slightly inclined orbit to keep the spacecraft safely out of the F ring. Once the periapse position is established, conceivably as little as half an orbit later, a maneuver at periapse would circularize the orbit between the F and A rings and bring its inclination to zero, almost ready for HOI. The two-step approach would start with a maneuver circularizing the orbit at the last pumpdown orbit's periapse, at that safe radius just outside of the F ring. This would be a "safe haven" position, an orbit that could be maintained for some time while checkouts ensure readiness for the final commitment. Step two would have a maneuver that establishes periapse between the F and A rings, as in the one-step approach, and a final maneuver to circularize and

zero the inclination there. The propulsive requirements for the two-step version of this transfer are given in Section 3, Mission Design—Pump Down V-infinity Leverage Tour Phase and HOI.

For either approach, one final event would have the spacecraft ready for HOI: jettisoning the now-spent SOI/pumpdown stage. The period that includes these final maneuvers and events of the SOI/pumpdown phase would be another operationally intense period, supporting much navigation work, the maneuvers, TCMs to prepare for and clean up those maneuvers, and spacecraft health monitoring and checkouts.

The spacecraft could remain in this ready-for-HOI orbit for some time without undue risk. Checkout of spacecraft systems, particularly the EP subsystem, could ensure readiness for the hover orbit. However, the "warranty clock" on lifetime-limiting components would be ticking, so there would be practical limits to available loiter time.

HOI is the moment the EP or chemical engines would turn on with the intent of traversing inward over the rings. From that point onward, with few exceptions, the EP engines *must* be running, or the chemical engines firing approximately 4 times per orbit, to prevent collision with the rings, so operations would include close monitoring of the propulsion subsystem health. Normal spacecraft safing procedures would not work here. Spacecraft safing events could require an on-call quick-response team and onboard autonomy. Safing *could not* be allowed to interrupt hover thrusting or allow attitude changes that make the hover thrusting ineffective. Some combination of inertial and/or sun-sensing attitude determination capability must be available to safing routines immediately, so that at least rough pointing of the hover thrust vector is maintained. Once that is established, the spacecraft must establish a rough direction to Earth and communicate with Earth, most likely through a medium-gain antenna (MGA) boresighted with the high-gain antenna (HGA).

Once over the outer edge of the A ring, the science observations would begin. Data would be compressed and stored on the spacecraft for daily downlink passes. Periods of traverse would be at relatively low average data rates, since there would be no attempt made to measure particle velocities and spin states. Full hover periods, when traverse would be stopped and the spacecraft flies in formation with the ring particles beneath, would be high-rate periods, with images taken sufficiently frequently to establish particle spin states and velocities before and after collisions. Data generated during these periods, even with compression, would be too voluminous to return in near-real-time. Near-real-time snapshots are possible, but the bulk of the full-hover data would be returned during the following traverse period. Depending on the size of the spacecraft's HGA and the number of 34-m receiving apertures arrayed, Ka-band downlink data rates could vary from less than 40 kbps for 35 W RF through a 2.5-m HGA to a single 34-m DSN station, to approximately half a Mbps for 50 W RF through a 4-m HGA to four arrayed 34-m apertures (approximately a 70-m equivalent). Even this most capable case falls short of returning all data from the most demanding of science scenarios; therefore, progress in telecommunication technologies that increase data rates from the outer solar system would be very useful, providing increased aperture on the ground, better data compression, etc.

The ability to detect and explore unexpected targets of opportunity in the rings, possible with EP traverse, raises the topic of onboard autonomous science planning and execution. If the operations team must be in the loop, the delay between the science instruments observing a target of opportunity and the reception of the commands at the spacecraft to return and observe the target would cause complexities with operations and demands on spacecraft resources such as unacceptably high propellant use. However, if the spacecraft has the onboard algorithms to sift through data being taken (notably, more data than could be downlinked) and detect, within a few seconds, the presence of a phenomenon of great interest, it could immediately invoke a sequence to stop the traverse and circularize the orbit at that spot, begin storing data in the full-hover mode, and notify the operations team this has occurred. Such algorithms would need careful design and choice of trigger parameters, since there would be the potential for improper choices to result in the algorithm triggering every few minutes, impeding the mission and quickly convincing the operations team to disable this autonomous feature.

During the hover orbit, the spacecraft and operations team would have to deal with ring collision hazards, some at well-known locations, others that would be surprises. The known hazards would require only command sequences to increase temporarily the spacecraft's offset distance, together with increasing the radial traverse rate, "hopping" to avoid the hazards. Many such hazards are longitudinally localized, so by timing the arrival at the radial position of such a hazard (by adjusting the traverse rate), the operations

team could arrange to have the hazard up to 180 degrees in longitude away from the spacecraft, avoiding the hazard altogether. Surprises would be more difficult operationally and might require another on-call quick response team in addition to autonomous spacecraft detection and reaction. Such a team would evaluate downlinked data concerning what triggered the spacecraft's hazard-avoidance response, determine the nature of the hazard, and in the short-term, recommend which of a set of onboard canned response sequences to initiate. This would be done while the spacecraft maintains a larger-than-usual offset, using propellant at a higher rate; therefore, a quick response would be important. Once clear of the hazard, the spacecraft could be put into its standard full-hover mode while the operations team does a more thorough assessment of the hazard and the appropriate long-term response.

The hopping function could be useful in another circumstance. Occasionally, the operations team might find it advantageous to hop (without increasing the traverse rate) to a higher offset, to get a farther-horizon view of what lies ahead. This would require imaging, most likely at WAC resolutions and fields of view, in a direction very different from the usual nadir view. The study team did not address the question of how to implement this, but example options would include a separate imaging instrument or a simple flip-mirror for the science WAC. The operations team would have to generate sequences, or supply parameters to a standard sequence, to perform a voluntary hop.

Saturn's rings also might present a hazard that is a contamination hazard rather than a collision hazard. The "spoke" phenomenon observed by both Voyager and Cassini is thought to involve electrically charged dust being elevated out of the mean ring plane by electrostatic fields. If so, charging of the spacecraft, if opposite in sign to the charge on the dust particles, might attract dust particles, causing them to adhere to a variety of surfaces, such as optical surfaces of science instruments and star trackers. If detected, this could be mitigated by intentionally charging the spacecraft to repel the particles. For the EP implementation this is a simple matter. The EP system continually operates a beam neutralizer, an electron gun that emits electrons into the ion beam to neutralize it, preventing unwanted spacecraft charging. Varying slightly the current from the neutralizers can tip the balance to yield a net spacecraft charge of the desired sign and intensity.

At EOM, there are no stressing operations requirements. Simply depleting the xenon propellant, or intentionally turning off the hover thrust, would naturally allow the spacecraft to settle permanently into the ring plane.

Mission Design

Before examining how to leave Earth, it is instructive to see how heavy a spacecraft might be needed at the start of the hover phase. Hovering above the rings of Saturn requires a net thrust to be applied to the spacecraft perpendicularly away from the ring plane. The thrust can be applied either continuously, or as continual pulses, spaced at most approximately two to four hours apart, depending on the radial distance from Saturn. For even the lowest tolerable altitude above the ring plane, this thrusting requirement results in a significant delta-V requirement and correspondingly large propellant mass. Thus, as shown earlier in the report, a reasonable spacecraft wet mass at HOI is in the range of 800–2,100 kg.

To achieve the sweet-spot mass at HOI with the smallest possible launch vehicle and a mission duration of 14–15 years, chemical and SEP trajectories to Saturn would be the best candidates, in both cases using gravity assists. Detailed NEP studies were excluded based on a preliminary orbit-energy analysis that showed NEP to be too heavy at current specific power levels, which would necessitate a larger launch vehicle. Solar sail and more immature propulsion technologies, such as mini-magnetospheric plasma propulsion (M2P2), were not considered on account of their low TRLs. SEP usage would be done as a separate stage, which would be dropped before SOI.

The HOI point, just outside the A ring at a Saturn range of approximately 139,000 km, is deep inside the gravity well of Saturn. Thus, significant changes in orbital energy would be required from SOI to HOI. Indeed, combining SOI and HOI into a single maneuver to insert directly into a circular, 139,000 km orbit costs at least 7 km/s, a prohibitive amount. Thus, a pumpdown strategy would be employed to go from SOI to HOI, involving a long series of flybys and small maneuvers, terminated by a large HOI maneuver after the last flyby. Flybys of Titan, Rhea, Dione, Tethys and Enceladus would be used; the other satellites would be too small or have inconvenient orbits to be of much benefit. Propulsion choices for the

pumpdown phase and HOI would be chemical and REP. As shown earlier in the report, the hover phase would be better accomplished with a REP system than with a chemical system, although both are viable. A REP hover system could also be used for the small maneuvers, with only small gravity losses incurred, but a chemical burn would be needed for the HOI. Replacing HOI with a REP spiral-down after the last flyby would not be worthwhile. It would add upwards of 2 years to the flight time for a 1-kW, 1000-s I_{SP} REP hover system and a 1000-kg spacecraft at hover start, and yet would provide only limited mass benefit because of the gravity losses and low sweet-spot I_{SP} (~ 1000 s) of the REP hover system. Higher I_{SP} would give greater mass benefit to the spiral-down, but cost even more flight time at 1 kW.

If NEP were used for the interplanetary trajectory, it could also be used during the pumpdown phase. It would have separate engines from the REP hover system because higher I_{SP} would be needed, particularly on the interplanetary portion, and because there would likely be too much throughput. Thus, the NEP would be dropped as a stage before hover start. NEP would enable neither a faster pumpdown phase nor a faster spiral-down to hover start. Thus, although it might provide some slight benefit because of the higher I_{SP} and higher power, using NEP in the pumpdown phase would certainly not provide enough benefit to allow launching on a smaller launch vehicle than the chemical or SEP missions require.

The thrust necessary to maintain the hover orbit has been analyzed by [14]. In the present study, for purposes of stability and of not perturbing the nadir (and main camera target) point on the rings with a thrust plume, a tripodal or similar thruster arrangement is anticipated. This would also allow simultaneous hovering and radial traversal of the rings by changing the spacecraft and/or thruster attitude to fire thrust components both perpendicular to the ring plane and against the spacecraft velocity direction.

Interplanetary Trajectories

Broad Search for Gravity-Assist Trajectories. A broad search for gravity-assist trajectories was first conducted by Damon Landau using a method and software he developed involving combinatorial searches over Lambert arcs between flyby bodies on discrete dates (Figures 3-6 and 3-7). Between two and four flybys of Venus, Earth, or Mars would be permitted, with the option of performing a Jupiter flyby thereafter. This method would provide the trajectory paths and rough flyby dates that would likely be the best candidates in a more refined search. The delta-V metric is fairly well correlated with the delta-V of optimized chemical trajectories, but at best only roughly correlated with the delta-V of optimized EP trajectories.

Optimized Gravity-Assist Chemical Trajectories. Selected trajectories from the broad search, as well as trajectories developed “by hand” were optimized in a medium-fidelity preliminary trajectory design tool, MALTO, which yields good correlation with high fidelity tools. Trajectory plots show representative trajectories; only deterministic delta-V is included, there are no launch period effects, and the arrival mass indicated is the spacecraft mass just prior to SOI.

Assessing the benefit of JGA trajectories is a complex undertaking, especially when circumstances drive the mission to long transfer durations. Of course, using Jupiter to get to Saturn requires that the two planets be properly aligned, which occurs for ~2-year windows on ~20-year centers. Projects must trade between the desire to use the advantage of a JGA trajectory and the need to fit programmatic schedules. But there are other trades to be made between, for example, mass delivered to Saturn approach and trip time. Some trades depend on what mission events or spacecraft capabilities are required after Saturn approach, adding another layer of complexity. For instance, one JGA trajectory might deliver more mass to Saturn approach than a particular non-JGA trajectory, but its V-infinity of approach to Saturn is significantly higher. If the spacecraft is to perform a propulsive SOI maneuver, the extra propellant mass needed for SOI from the higher approach V-infinity might offset the larger approach mass.

Increasing the mass a given launch vehicle can deliver to an initial trajectory is a function of launch C3. Launching directly from Earth to Jupiter is a quick way to Jupiter, approximately 2 years, but the C3 of 80 km^2/s^2 reduces the mass capability of an Atlas V 551 to less than 1 ton. Launching to a 2-year Earth return trajectory, for a Delta-VEGA assist to Jupiter, reduces the C3 to ~25 km^2/s^2, greatly increasing the launch mass, but also requiring that some of that mass be spent on propellant for the sizeable deep space maneuver (DSM) that is an inherent part of Delta-VEGA trajectories. Launching on a Venus flyby trajectory for a Venus gravity assist reduces C3 to as little as 9 km^2/s^2 and requires no DSM, but there are associated complications. A single gravity assist at Venus after a transfer from Earth is not sufficient to

get to Jupiter; additional inner solar system gravity assists are needed, increasing the transfer duration. Also, Earth, Venus, Jupiter, and any other body participating in the gravity assist scheme must be in the proper positions. “Ballistic” trajectories that use such gravity assists without the need for a DSM do not recur frequently, and sometimes the mass performance is reduced by the need to launch to a higher C3 to overcome non-optimal alignments of the planets involved. Such sub-optimal performance can be improved by introducing DSMs, but of course some of the increased mass capability is spent on propellant for the DSMs.

Currently, the method for achieving the largest launch mass capability for a JGA trajectory is to launch to a C3 barely greater than 0, and to use SEP to set up inner solar system gravity assists leading to a JGA. This is beneficial if the mass augmentation from decreasing C3 to 0+ is larger than the wet mass of the SEP system. This is usually the case for envisioned SEP systems using, for instance, NEXT ion engines and Ultraflex solar arrays.

One result of using these increasingly complex JGA trajectories to increase the delivered mass is generally increased trip times. But the increase in delivered mass per increase in trip time falls rapidly after approximately 10–12 years, so the mass capability of gravity assist trajectories to Saturn that use only inner solar system planets can catch up to, sometimes even surpass, JGA trajectories of equal trip time. This seems to be the case with the trajectories found by this study’s broad search. Despite this general trend, there are occasional “sweet spot” trajectories that give JGA an edge, typically in reduced trip time while maintaining delivered mass capability. Finding these sweet spots can be distinctly *not* straightforward, despite tools based on such recent advances as Tisserand graphs [17], and can take significantly more time than was available for this study. In future studies further, more comprehensive searches for JGA trajectories would be prudent.

As mentioned above, launch mass capability is only one factor in the mass actually delivered to do the science mission. V-infinity and declination of the Saturn approach, maneuvering within the Saturn system to get to HOI, all these influence the mass finally delivered to HOI, and must be considered along with the performance of JGA and non-JGA trajectories.

Opportunities for JGA trajectories to Saturn center around Jupiter flybys in late 2000 (Cassini used this one), 2021, 2041, 2060, etc. An SRO mission with a project start after 2022 could use a Jupiter flyby around the year 2041 to increase the arrival mass or decrease the flight time. A sample trajectory is shown in Figure 3-6. The Jupiter flyby date window is a better metric than a launch date window because, as discussed above, there are many different trajectory types, with widely varying trip times, to get from Earth to Jupiter. Potential launch dates could be as little as 3 to 3.5 years before the Jupiter flyby date, or as much as 8 years before, limited by total mission duration. The shorter flight times tend to emphasize trip time reduction with respect to non-JGA trajectories, but at the expense of smaller delivered mass. With JGA, lowering the arrival V-infinity to less than approximately 3.8 km/s would require a great increase in time of flight (TOF) or great reduction in arrival mass, relative to that shown in Figure 3-8. Without JGA, lowering the arrival V-infinity to less than approximately 5.2 km/s would result in great reduction in arrival mass, relative to that shown in Figures 3-9 and 3-10.

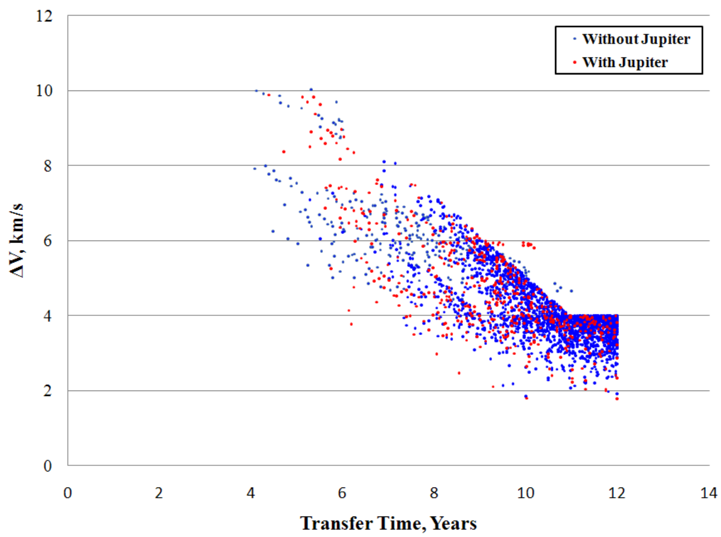

Figure 3-6. Broad Search: Delta-V Versus Transfer Time

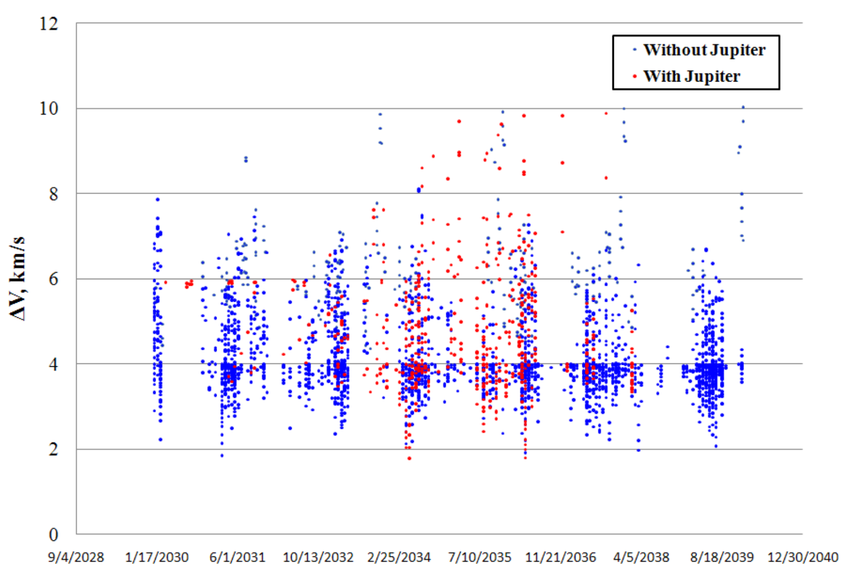

Figure 3-7. Broad Search: Delta-V Versus Launch Date

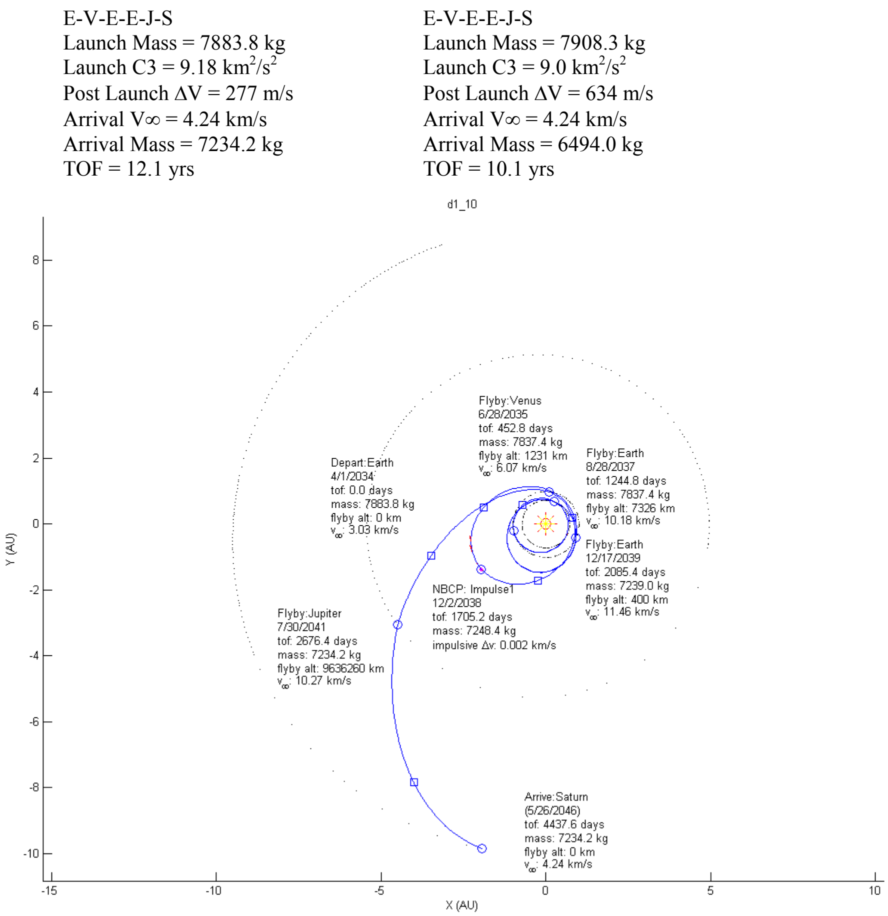

Figure 3-8. Sample Chemical Trajectory with Jupiter Gravity Assist, Delta IV Heavy

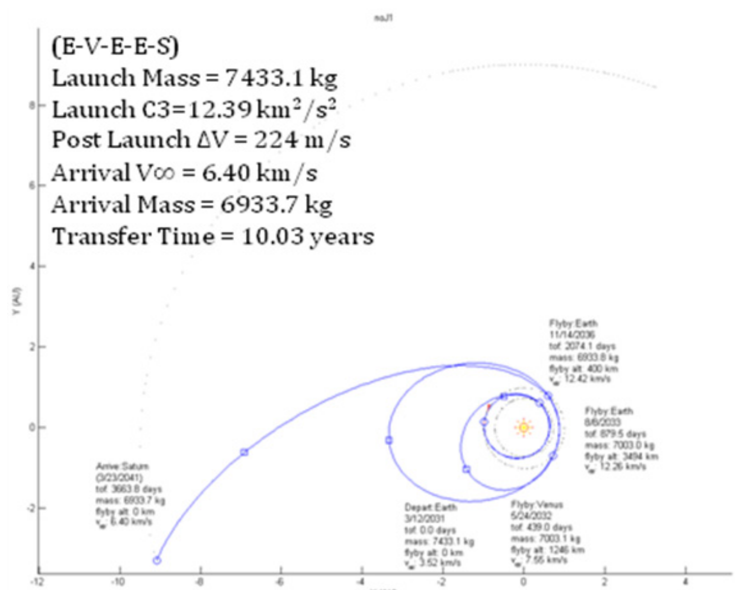

Figure 3-9. Sample Chemical Trajectory without Jupiter Gravity Assist, Short TOF, Delta IV Heavy

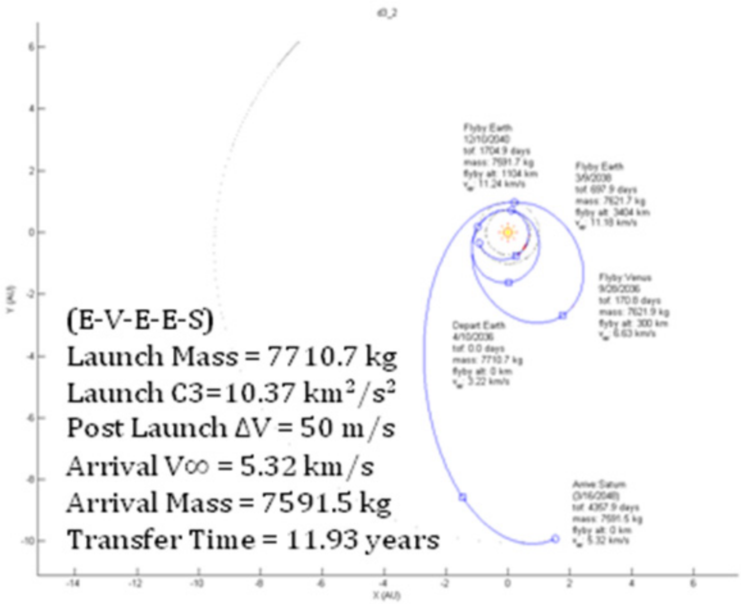

Figure 3-10. Sample Chemical Trajectory without Jupiter Gravity Assist, Longer TOF, Delta IV Heavy

Optimized Gravity-Assist SEP Trajectories. Jupiter is in position to provide similar assistance as it did in the chemical cases. With the SEP, arrival V-infinities are generally lower than in the chemical case. Representative trajectory plots are shown in Figures 3-11 and 3-12 with and without JGA. The indicated arrival mass includes the SEP stage mass and is prior to SOI, as in the chemical case. For the case with JGA, lowering the arrival V-infinity to approximately 2.5 km/s would be easily achievable with approximately two extra years of flight time. In non-Jupiter cases, it is generally very difficult to reduce the arrival V-infinity below approximately 5 km/s. Array reference power, P_0, is taken as 15 kW, and 2+1 NEXT thrusters are used (two primary, one backup).

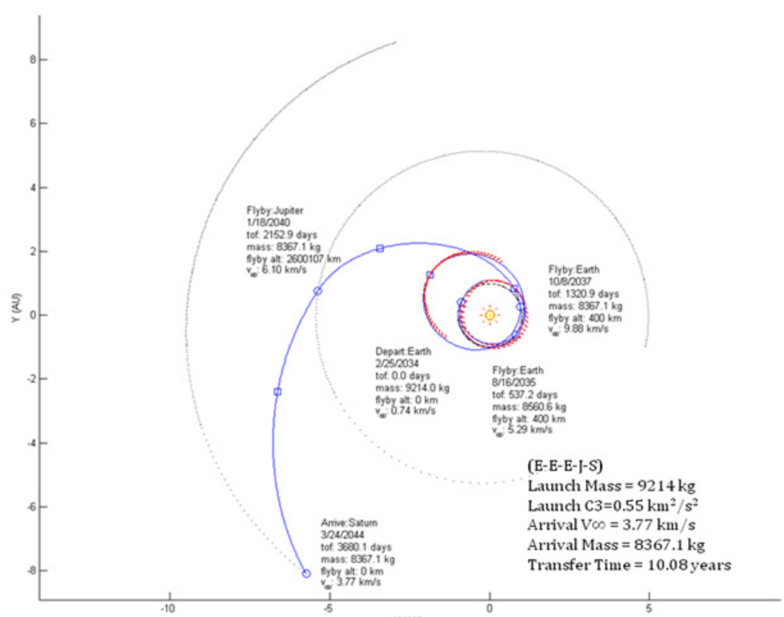

Figure 3-11. SEP with Jupiter Gravity Assist, Delta IV Heavy

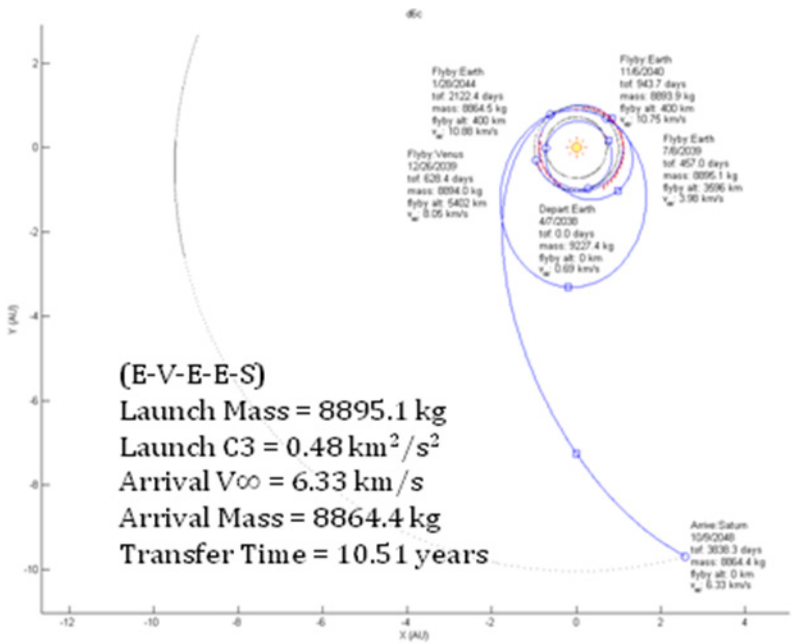

Figure 3-12. SEP without Jupiter Gravity Assist, Delta IV Heavy

Augmenting the Interplanetary Trajectories with REP. One of the chemical trajectories and one of the SEP trajectories were examined with REP on the final leg before Saturn. A 1-kW, 45%-efficient system was considered, with a maximum I_{SP} of 1,500 s. As can be seen in Tables 3-12 and 3-13, it is very costly in terms of mass to reduce the arrival V-infinity or the flight time. The mass hit is smaller if taken at SOI.

Table 3-12. Electric Propulsion with REP Augment

Arrival V-infinity (km/s)	Mass (kg)	Additional Propellant (kg)
3.77	8,367	0
3.5	8,286	81
3	8,083	284
2.5	7,830	537
2	7,521	846

Table 3-13. Chemical Propulsion with REP Augment

Thruster Efficiency	Arrival V-infinity (km/s)	Mass (kg)	Additional Propellant (kg)	TOF to SOI (yrs)
0.45	6.4	6,933	0	10.03
0.45	6	6,782	151	10.23
0.45	5	6,342	591	10.59
0.45	4	5,546	1387	11.11
0.45	3	4,675	2258	11.66
0.55	3	4,976	1958	11.69
0.65	3	5,204	1729	11.70

SOI: Joining the Interplanetary and the Saturn Phases

Performing SOI low in the gravity well of Saturn can save significant amounts of delta-V, especially for the higher arrival V-infinities, as shown in Figures 3-13 to 3-15. However, the rings might pose a barrier to performing SOI at low radii. Fortunately, good arrival geometry at Saturn would generally be available in the 2044–2050 timeframe, which encompasses the arrival dates considered in this study. Outside of these dates, the normal arrival V-infinity declination would be such that it is not possible, without piercing the rings, to place periapsis (and the SOI location) in the ring plane between the innermost ring and Saturn's atmosphere. This situation would impose a delta-V penalty, either to adjust the declination, or to perform SOI at a larger radius.

The nominal SOI strategy for an EEEJS SEP trajectory is as follows:

1. Perform SOI at 1.1R_s (chemical), budget: 350 m/s
2. Change inclination at apoapsis to target Titan flyby (REP), budget: 100 m/s
3. Use Titan flyby to raise periapsis above rings

The delta-Vs and orbits associated with various SOI strategies are shown in Figures 3-13 through 3-16.

Changes to the nominal strategy would be made for different arrival V-infinities and geometries as follows: delta-V cost goes up considerably for arrival V-infinity beyond approximately 4 km/s, or poor geometries (although typical arrivals in 2044–2050 have geometries that add only small additional delta-V cost for declination problems).

An alternative SOI strategy would be to use a Titan AGA on approach to Saturn. This would leave the spacecraft in a captured Saturn-centric orbit, with apoapsis somewhat beyond Titan's orbit and periapsis a little below Enceladus's orbit. The Titan AGA would remove the need for SOI and the inclination change maneuver (or periapsis raise maneuver). Casoliva and Lyons [18] show that arrival V-infinities at Saturn of up to approximately 6 km/s are tenable for capturing using Titan AGAs, well above the V-infinities expected.

After SOI, the pumpdown phase would begin.

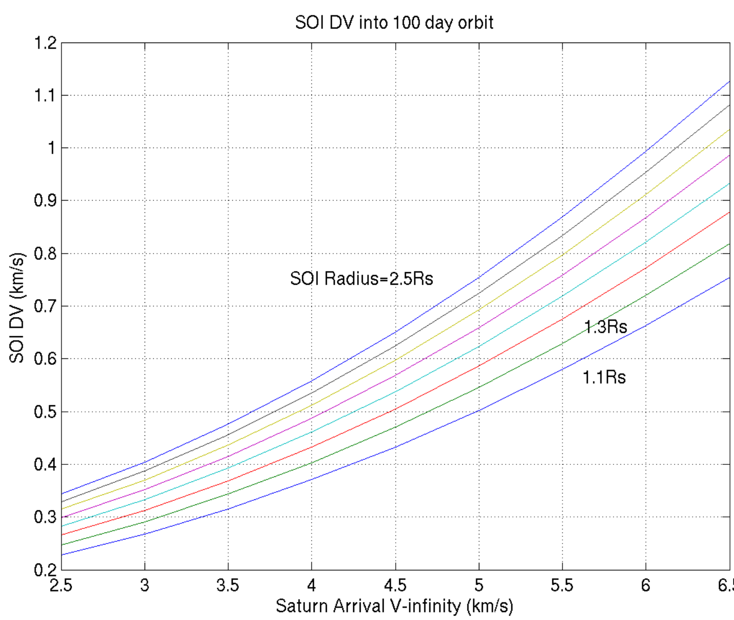

Figure 3-13. Impulsive Capture Delta-V into Various Saturn Orbits

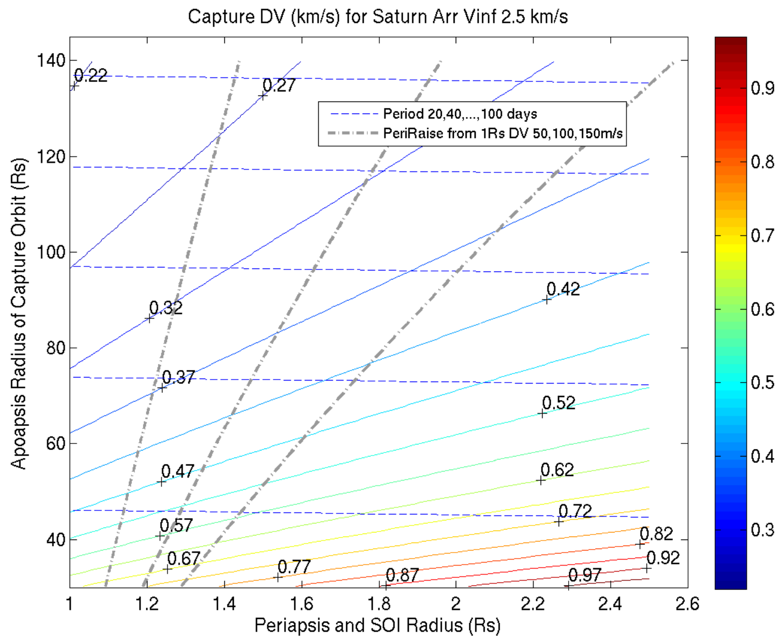

Figure 3-14. Capture Delta-V for Various Orbits and Effect of Applying Delta-V at Apoapsis; Arrival V-Infinity of 2.5 km/s

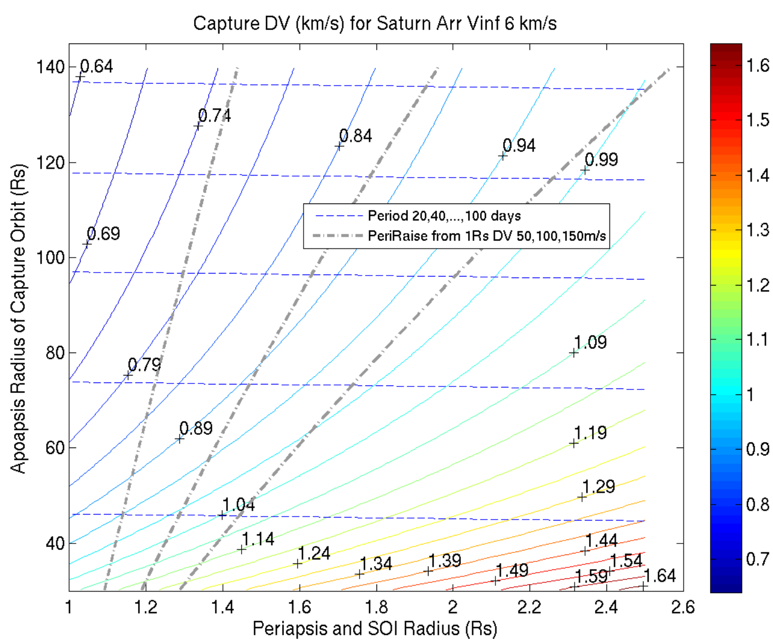

Figure 3-15. Capture Delta-V for Various Orbits and Effect of Applying Delta-V at Apoapsis; Arrival V-Infinity of 6 km/s

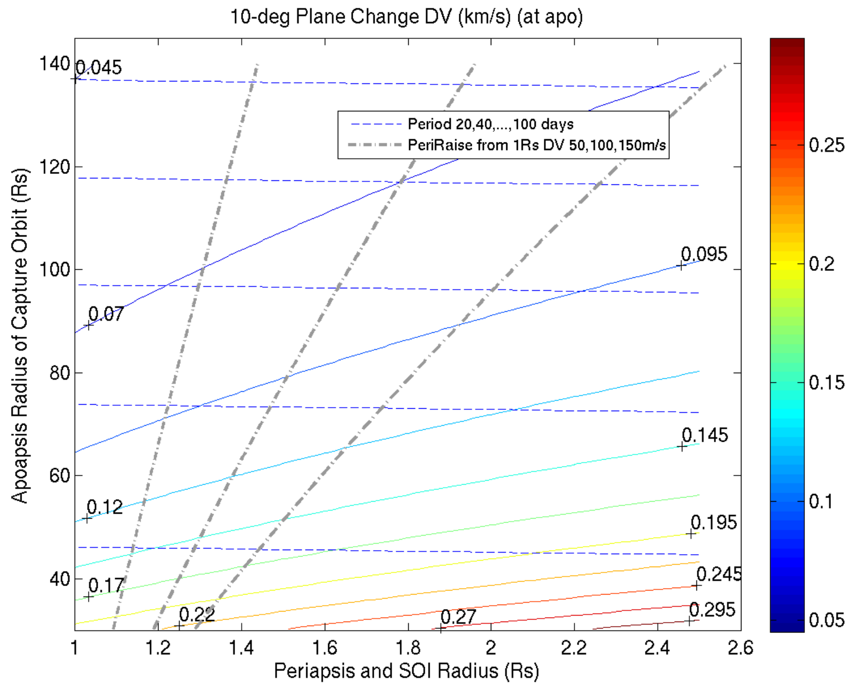

Figure 3-16. Plane Change Delta-V, Useful for Targeting the First Titan Flyby of the Pumpdown Phase

Pumpdown, V-infinity Leveraging Tour Phase, and HOI. The sequence of events and associated delta-Vs for this phase are as follows:

- Approximately 70–80 flybys of Titan, Rhea, Dione, Tethys, Enceladus
- Places spacecraft into orbit with periapsis at 142,000 km, apoapsis at Enceladus
- Deterministic leveraging delta-Vs and flight time SOI-HOI, options:
  - ~330 m/s chemical, 3.5 years
  - ~400 m/s REP 1 kW, 3.6 years
  - ~460 m/s REP 0.5 kW, 3.6–4 years
- HOI-1 (circularize outside F ring): 2.95 km/s, chemical
- HOI-2a plus HOI-2b (F ring hop): 230 m/s
- Statistical delta-V: ~100–200 m/s (can be done using REP)

An alternative, performing HOI as a single burn inside the F ring does not save significant delta-V and is a riskier strategy. Another alternative, using REP to spiral in after the last Enceladus flyby and hence to reduce HOI-1 delta-V takes too much flight time for significant delta-V reduction.

The design of the pumpdown phase is based on the Tisserand graph method and on the detailed analysis presented by Campagnola, Strange, and Russell [19] for the pumpdown phase of an Enceladus orbiter mission. Since the goal here is not to enter orbit around Enceladus, but to circularize just outside Saturn's rings, the pumpdown phase is rather different from the orbiter pumpdown phase in that the spacecraft flies by a number of satellites before abandoning the most distant one, rather than flying by one satellite repeatedly and then leaving it to flyby the next satellite repeatedly. A Tisserand graph for the moons and orbits of interest is shown in Figure 3-17. A representative sequence of orbits around Saturn is shown in two parts in Figures 3-18 and 3-19.

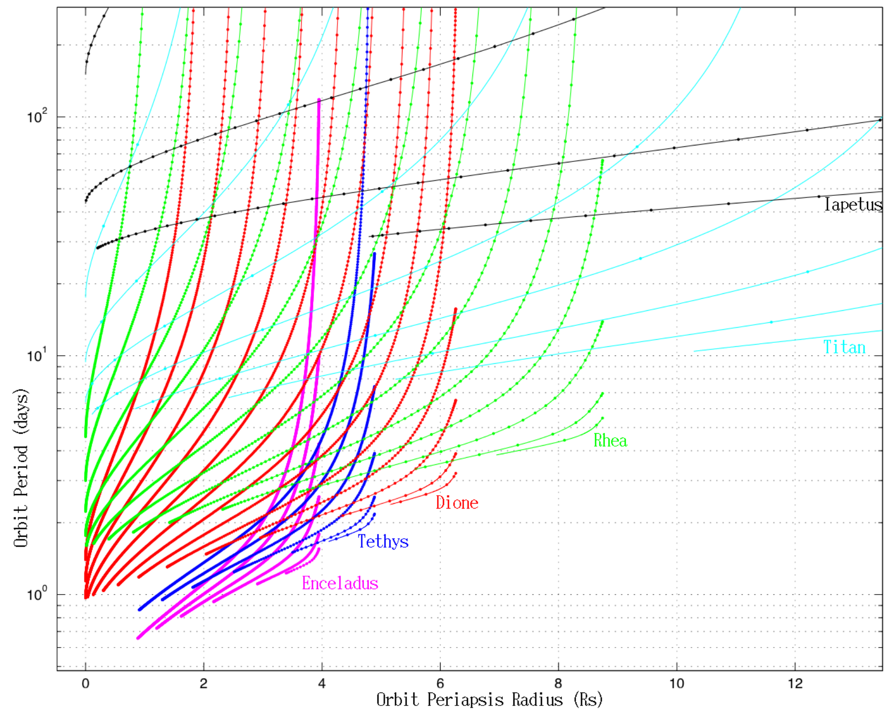

Figure 3-17. Tisserand Graph. V-infinity contours for Enceladus, Tethys, Dione, Rhea, Titan, Iapetus, at values of (0.5, 1, 2, 3, ...) km/s. Distance between dots represents a 100-km flyby.

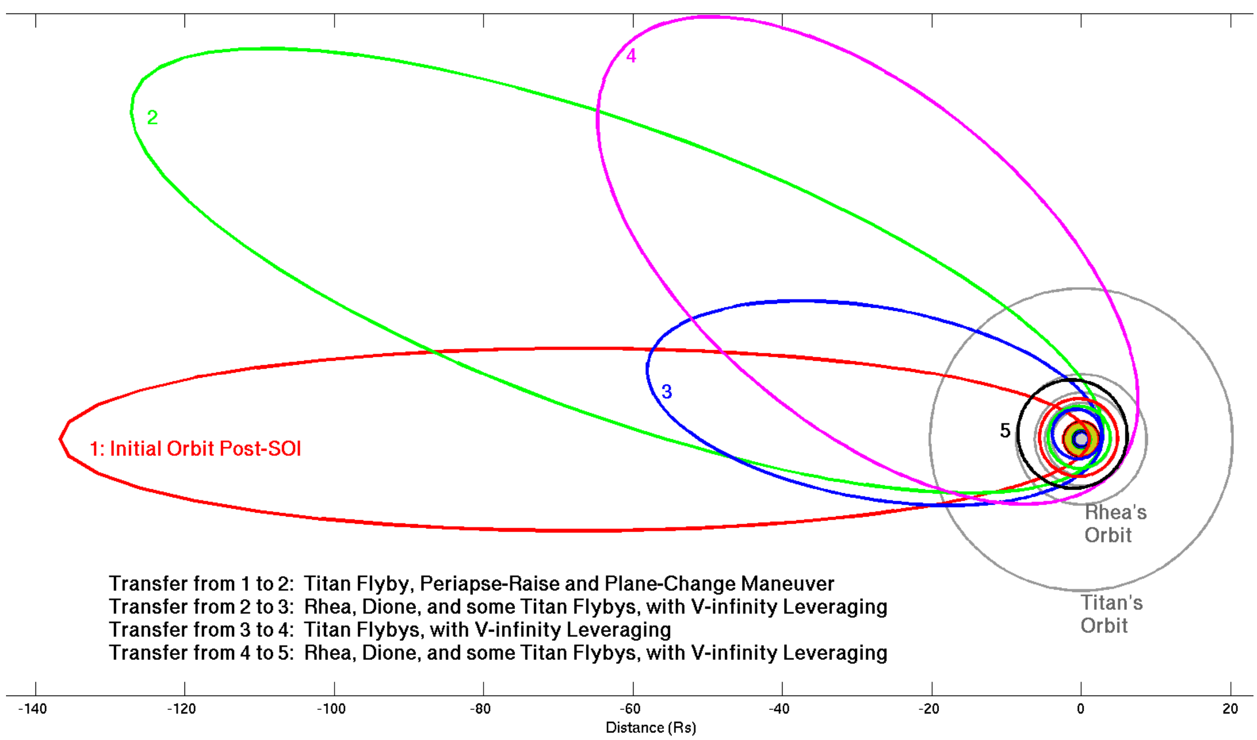

Figure 3-18. Sample Orbits in Pumpdown Phase—First Part

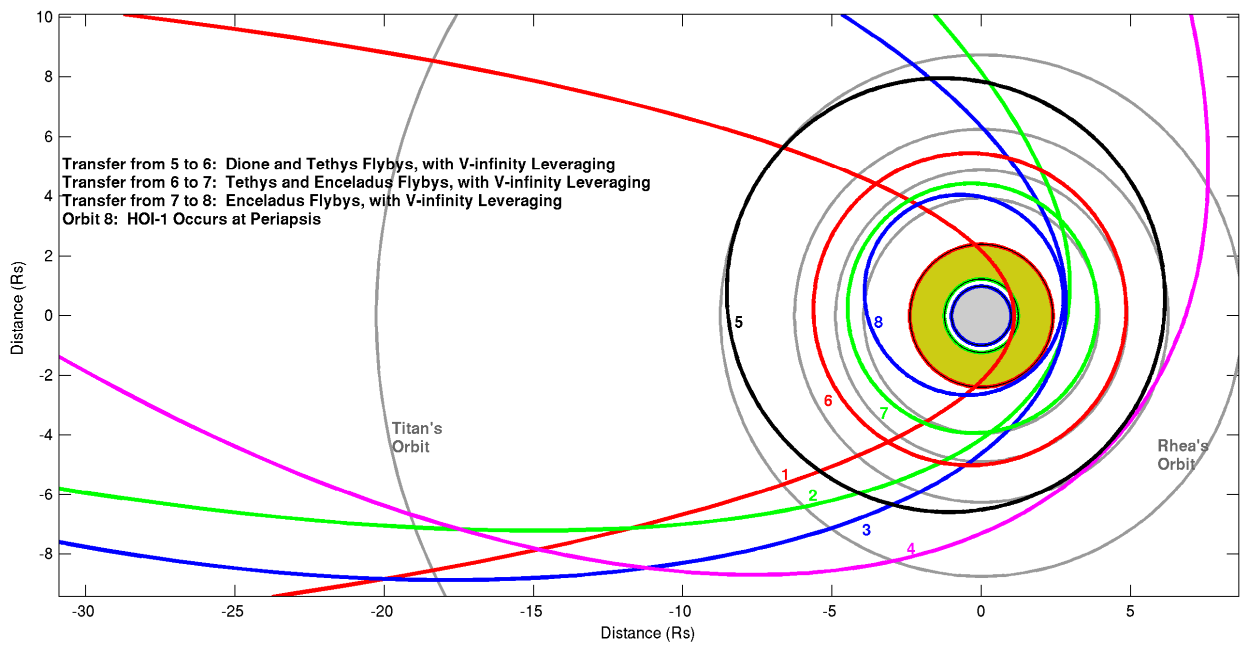

Figure 3-19. Sample Orbits in Pumpdown Phase—Last Part

HOI (i.e., insertion into the orbit in which the spacecraft would be when the hover system is activated) could occur as one or three main burns. In both cases, some inclination would have to be removed because of the need to avoid the F ring, which, because of its inclination and nebulousness, extends approximately 25 km above and below the ring plane in maximum extent. The one-burn strategy is the riskier of the two as it would circularize an orbit that has periapsis already inside the F ring and apoapsis at Enceladus's orbit. The three-burn strategy would first circularize just outside the F ring, then perform two burns to hop over the F ring and circularize outside the A ring. The hover phase could then begin.

This pumpdown strategy is more robust to missed encounters than the Cassini Saturn tour. No analyses were done to verify the ability to resume the pumpdown after a missed encounter, but the cost of such a failure would most likely be a few m/s of delta-V and one to a few months to re-establish proper phasing [Strange, private communication]. Spending more delta-V could accelerate this schedule.

Hover Phase. HOI, whether done as one or three main maneuvers, puts the spacecraft in a circular orbit just outside the A ring. The SOI pumpdown stage would be jettisoned at this point. After the health of the spacecraft is confirmed, hovering and radial traverse could commence.

The hovering problem was studied by [14]. With continuous thrust, the axial thrust acceleration needed for hovering would be approximately proportional to the hover height above the ring plane, and inversely proportional to the cube of the radius from Saturn. A hover height of 2 km is assumed as being sufficiently safe from ring hazards and flight path control anomalies, and close enough for science purposes and requiring a reasonable thrust acceleration level. The thrust would be provided by a number of thrusters firing simultaneously. With the appropriate thruster configuration and gimballing, a variety of benefits would be achieved. The disturbance by the thruster plumes of the ring particles in the nadir direction could be reduced, attitude control functions could be performed, and simultaneous hovering and radial traverse could be more easily achieved.

When propellant runs out for hovering, the most likely disposition strategy for the spacecraft would be to let it impact the rings. It is unlikely that the mission would be able to guide the spacecraft to a safe spot within the rings (either in one of the divisions or between particles). The final unpowered trajectory arc of the spacecraft may itself be chosen so as to contribute to the science of the mission in some fashion.

It is essential to note that the spacecraft must be able to control its thrusters autonomously to maintain its altitude above the ring plane, as well as its attitude. There is no time for ground-in-the-loop control. This would require the development of new control algorithms, and the refinement or development of new sensors to provide input to the control algorithms.

Planetary Protection

NASA's Planetary Protection Officer has not categorized Saturn's rings because they appear to be an unlikely location for pre-biological or biological activity; the vacuum condition and extremely low temperatures preclude any form of liquid water. Any planetary protection issues associated with this mission would arise from the possibility of accidental collision with Titan or Enceladus en route to the ring hover orbit, whether the hover spacecraft would be powered by RPSs or a nuclear fission reactor power system. This study assumed that analyses would show the probability of contaminating bodies of liquid water at either Titan or Enceladus as less than 10^{-4}, and did not address the requirements that would be levied on the project and spacecraft if this were not the case. Such collisions could occur only before the initiation of the hover orbit; once in the near vicinity of the rings, it would be exceedingly difficult for the spacecraft to escape to the higher orbits needed to collide with either Enceladus or Titan. The age of Saturn's rings, estimated to be between hundreds of millions to multiple billions of years, is testimony to the rarity of objects the size of ring particles (which includes the size of the spacecraft) being ejected from the rings.

This fact leads to the spacecraft disposal scenario. At EOM, termination of the hover thrust, either accidentally, by design, or by simple depletion of propellants, would have the spacecraft naturally settle into the ring plane and become just another of the many billions of ring particles.

Another aspect of planetary protection is the protection of Earth. Any potential issues of this kind would arise early in the mission, since the mission would return no extraterrestrial materials to Earth. Early in the mission, there could be Earth gravity assist flybys. Mission designs that involve RPSs or operation of nuclear fission reactor power systems and Earth flybys for gravity assists (or for any other reason) would need to use approved operations plans, such as aim point biasing during Earth approaches.

Risk List

This study considered technology development risks as an integral part of the technology assessment effort. These risks are discussed earlier in the Technology Description section. This study did not perform a detailed consideration of mission risks, however.

Nevertheless, during discussion of SRO mission options, a number of mission and implementation risks were discussed as they relate to mission architecture options and their prospects for success in achieving the science objectives. These risks were not quantified as to likelihood or consequence of occurrence, and the list below cannot be considered exhaustive.

Mission Risks

1. Spacecraft failure due to component reliability needed for long mission lifetime (~15 years)
2. Achieving acceptable initial Saturn orbit when using AGA at Titan
3. Adequacy of spacecraft autonomous sensing and control algorithms for maintaining separation from the ring plane
4. For options using Saturn aerocapture, small time separation between critical events (in this case, there would only be ~3.5 hours between aerocapture and HOI)
5. Proper orbit state for HOI not achieved; large maneuver near dangerous zone (ring particles)
6. Ion engine thrusters could interfere with dynamics of small ring particles, biasing science results
7. Complex hazard avoidance (navigation required to avoid ring hazards); see Appendix C.
8. Planetary protection requirements might preclude use of Earth gravity assist

9. Malfunction or underperformance of a very large bipropellant stage (SOI/pumpdown stage) that does a large number of maneuvers

Implementation Risks

1. Unavailability of plutonium
2. Reactor development program is not approved
3. Investment in reliability and long-life testing for power and propulsion technologies
4. Integration of potentially large number of ASRGs
5. Inability to demonstrate 10^{-4} probability of contaminating bodies of liquid water at either Titan or Enceladus, significantly increasing planetary protection costs
6. Investment in design for NEP radiation protection
7. Unavailability of appropriate thermal protection system materials for aerocapture or aerogravity assist

4. Development Schedule and Schedule Constraints

High-Level Mission Schedule

This study was primarily a technology study, not a mission study; therefore, schedules for the mission development phase (Phases A–D) were outside the scope of this study. However, mission timeline information for the operations phase (Phase E) has been provided in Section 2, Overview, and Section 3, Mission Design, for the discussion and graphical display of mission operations timeline information.

Technology Development Plan

This study identified candidate technologies and levels of performance. Since there is not a specific mission, with associated launch dates, the need dates are outside the scope of this study. However, technology development timeframes are identified below for many of the technology options discussed in earlier sections of this report. These technology developments are organized by science level to align technology needs with candidate mission science levels.

Achieving any of the four designated science levels would require developing the hardware, algorithms, and processes associated with the operations technologies that allow navigating and controlling the hover orbit. When this study team was being assembled, it was not anticipated that operations would assume such an important role. Though there was significant discussion of what operations technologies might be needed, the study did not analyze operations technologies in detail or generate schedules for development of those technologies. This is certainly a topic to be addressed in future SRO studies.

References to launch vehicles in the following discussions are based on simple, low-fidelity models of spacecraft mass and performance, and thus are not to be construed as definitive. Higher-fidelity studies are needed to confirm any such results.

Science Level 1

Level 1 science could be achieved using only current-technology chemical propulsion, even for the hover orbit, as described in Spilker [14]. Using a bipropellant hover-and-traverse propulsion system *might* keep the combined mass of the hover spacecraft and the SOI/pumpdown stage sufficiently small to launch on an Atlas V 551 without SEP, but verifying this would require a more detailed design study. Launching on a SEP-augmented Atlas V delivers sufficient mass that the hover spacecraft could use a monopropellant system for the half-year science mission. This would use a very different type of science orbit, not maintaining a fixed offset distance from the ring plane, and the traverse would be done in discrete, short-duration jumps rather than continuously. This is not amenable to detecting and stopping at unanticipated areas of interest. The lower I_{SP}, in conjunction with limits to the mass delivered to Saturn (for a given launch vehicle), limits the science mission duration, but because traverse is done in brief jumps, a far greater fraction of that period is spent in the hover-only mode, observing the behavior of the rings.

This could be done fairly easily with REP for hover and traverse, using the ASRG currently under development for electric power, and a technology development program that produces high-efficiency, low-power Hall thrusters. The project development schedule must include the development schedule for those Hall thrusters. This flight system might be launched to Saturn on an Atlas V without SEP.

If ^{238}Pu is not available for RPS power, a flight system based on a thermoelectric nuclear fission reactor power source and current-technology Hall or ion EP thrusters could perform level 1 science. Such a power source was the subject of a PSDS design study; results are reported in [13]. The power required for the NEP SRO mission concepts evaluated in this study are at the very low extreme for nuclear systems, resulting in the nuclear fission power systems suffering from low specific power in that range,

and not being a good match for the needs of an SRO mission. At level 1 there is no advantage to using Stirling conversion instead of thermoelectric. If the NEP solution were adopted, the flight system would require a SEP-augmented Delta IV Heavy for launch and transfer to Saturn.

Science Level 2

Level 2 science could be done with all chemical propulsion, including the hover orbit, as described in Spilker [14]. Using a bipropellant hover-and-traverse propulsion system might allow launching on a SEP-augmented Atlas V for a half-year science mission. The hover orbit would be similar in character to that of the level 1 science mission using chemical propulsion, but would traverse the much larger radial range of level 2. Assuming an appropriate SEP stage or kit has not already been developed, the project development schedule would have to include the development schedule for the SEP system.

The same type of REP hover-and-traverse propulsion system that achieves level 1 science, with more propellant and a relatively small increase in electric power, could achieve level 2 science, while still launching on an Atlas V without SEP augmentation. Again, the project development schedule must include the development schedule for the high-efficiency, low-power Hall thrusters.

It appears feasible to use NEP to achieve science level 2, but not with thermoelectric conversion. The estimated mass of such a system is greater than can be delivered to HOI by the “maximal delivery system” considered: the combination of a Delta IV Heavy launch vehicle, SEP augmentation, *and* either a Titan gravity assist or high-I_{SP} propulsion for SOI/pumpdown. But that combination could deliver a Stirling-based NEP system. At the power level needed for a NEP implementation of level 2 science (2–3 kWe), the Stirling system is lighter than the thermoelectric system.

Science Level 3

Science level 3 is the first big discriminator for technologies capable of satisfying science needs. All of the NEP and chemical implementation options fail to converge, by large margins, on that maximal delivery system: Delta IV Heavy, SEP augmentation, and either a Titan gravity assist or high-I_{SP} propulsion for SOI/pumpdown. The models indicate that at this level an ASRG-based system falls just short of fitting onto an Atlas V with SEP: to reach the 100,000 km radial position the required mass at HOI is ~20% over that system’s delivery capacity, or at the delivery capacity the traverse could reach only 103,000 km. But the margin there is sufficiently small that higher-fidelity analyses would be prudent. The formal result of the models is that at level 3, technology development is needed to produce RPSs with higher specific power than the ASRG. The SRG-550 easily meets the requirement, but at 550 We per unit the quantization mass penalty can be significant if the required power is slightly more than an integer multiple of 550 We. For this reason, development of the SRG-160, also called the “optimized ASRG,” would complement the SRG-550 and add significant flexibility in configuring power systems. A project’s schedule would need to consider technology development schedules for the SRG-550 and SRG-160, 7 and 5 years respectively. However, at the power levels needed for level 3, the efficiencies of existing Hall and ion thrusters are sufficient, so no additional technology development is needed. Technology development to increase the efficiency of medium-power EP thrusters would help in an effort to get an ASRG-based system onto a SEP-augmented Atlas V.

Science Level 4

This level is the most demanding, requiring more than the lift capacity of the maximal delivery system described above, even for the combination of the best power and best existing EP thruster systems considered. It might be possible to fit a level 4 system on the maximal delivery system by augmenting the hover spacecraft with higher-efficiency EP thrusters, but there appears to be a practical limit of about 60% on that efficiency, so the scope of mass savings from that approach is limited. Further mass reductions might be realized by using both Titan AGA and high-I_{SP} propulsion for the remaining pumpdown. The resulting list of technology developments needed for a level 4 science mission would be an impressive list of technologies to be developed for any single mission. The risks associated with all of these technologies needing to succeed in a timely manner make this level of science for a potential SRO mission highly challenging.

5. Integrated Assessment and Conclusions

The SRO study yielded some surprising results, not all of them concerning the technologies. The original intent for the study was to focus on electric power and in-space propulsion technologies, paying attention to the closely associated trajectories. However, technology needs also emerged in the general areas of operations technologies and telecommunications.

The principal high-level science value metric for the SRO mission turns out to be the radial range covered during the mission. The larger this traverse range, the larger the range of ring areal densities encountered, phenomena such as different types of waves observed, etc., and thus the higher the science value of the mission. Analyses indicate that the mission architectures most likely to be practical have the hover orbit starting at a Saturn-centric radius of ~139,000 km, between the F ring and the outer edge of the A ring, and traversing inward. The radius at EOM then gives the traverse range. The science team identified four science value levels, defined by the final traverse radius, as shown in Table 5-1. Level 1 is the lowest science value level and is considered the science floor or the minimum traverse needed to make the mission worth flying. At the other end of the spectrum, level 4 is the highest science value level and covers all regions and known phenomena of interest to the science team. At each level, the overall science addressed includes the science listed for that level, plus the science from all lower levels.

Table 5-1. Definition of the Four Science Value Levels, with Science Addressed in Each

Science Value Level	Final Radius (km)	Science Addressed
1	130,000	A ring—particle collisions, shapes, and spin; shepherded edges; waves; “propellers”; and self-gravity wakes
2	120,000	Denser A ring regions—viscous over-stability, wake-free region, and Cassini Division (low density)
3	100,000	B ring—very high-density regions and associated phenomena
4	84,000	C ring—“pedestal” region and small particle sizes

A critical technology that surprised the team by emerging from outside the power, propulsion, and trajectories areas, involves operations technology—the ability of the spacecraft to autonomously measure its distance from nebulous collections of ring particles, and then infer the distance to the mean ring plane (Saturn’s equatorial plane), for spacecraft navigation and orbit maintenance purposes. The need for this technology applies to all implementation options at all science levels. The development scope includes both hardware and software; no existing instrument appears capable of the measurements needed, and the algorithms for reducing the data on the spacecraft to yield the needed inferences do not exist. In some cases, the level of effort needed for these developments might outstrip that needed for the power and propulsion systems. There are potential science benefits from these kinds of measurements, especially if the system can measure individual ring particles’ out-of-plane velocity components.

Not surprisingly, the technology development effort needed for power and propulsion technologies to perform the hover orbit is generally highest for level 4 science and decreases with decreasing science level. However, this trend is not evenly distributed among power, propulsion, and operations technologies. If the mission is to launch on a Delta IV Heavy or smaller launch vehicle, even with SEP augmentation, fully attaining level 4 science would require significant advances in power technology. As shown in Figure 5-1, the development effort would have to produce power sources of specific power greater than that anticipated from the radioisotope-powered SRG-550, the most advanced generator concept considered. However, EP engines of power, thrust, I_{SP}, and efficiency levels appropriate to the Level 4 mission hover orbit already exist; therefore, no specific development is required in the propulsion area. Descoping science to levels 1 or 2, Figure 5-1 shows the specific power of the ASRG currently under development is sufficient. But the power and thrust levels of EP engines appropriate to the mission

profile are significantly smaller than the efficient operating ranges of current engines. Current engines throttled down to those ranges have efficiencies less than 30% to as little as 15%. Technology development would be needed to provide smaller engines, such as Hall thrusters with thrusts in the 20 to 50 mN range with efficiencies ~40% or better. Therefore, at the lower science value levels, the hover orbit technology emphasis shifts to propulsion technology.

Some EP technologies examined did not rate as highly as Hall or ion thrusters for the hover orbit application. Colloid thrusters might be useful in the future due to their high efficiency, *if* their I_{SP} can be brought up to levels near or above ~1,000 s and their reliability can approximate that of Hall thrusters. The best currently proposed designs have I_{SP} between 600 and 700 s, sufficiently low that it has a noticeable impact on the required propellant load, especially for the higher-level missions. Arcjet thrusters were rejected because they cannot run continuously for long periods.

One additional power technology considered for the hover orbit is fission-reactor-based power. Power levels considered are in the 1 to 3 kWe range. Current designs for such reactor systems cannot compete with Stirling radioisotope systems for high specific power. Figure 5-1 shows that SRO missions using reactor-based systems could launch only on a Delta IV Heavy with SEP augmentation, and even then could only barely achieve level 1 or 2 science. However, it is worthwhile to consider reactor power systems, because the availability of ^{238}Pu for future RPS-based systems is *not* assured. If none is available, reactor-based power would be the next EP alternative.

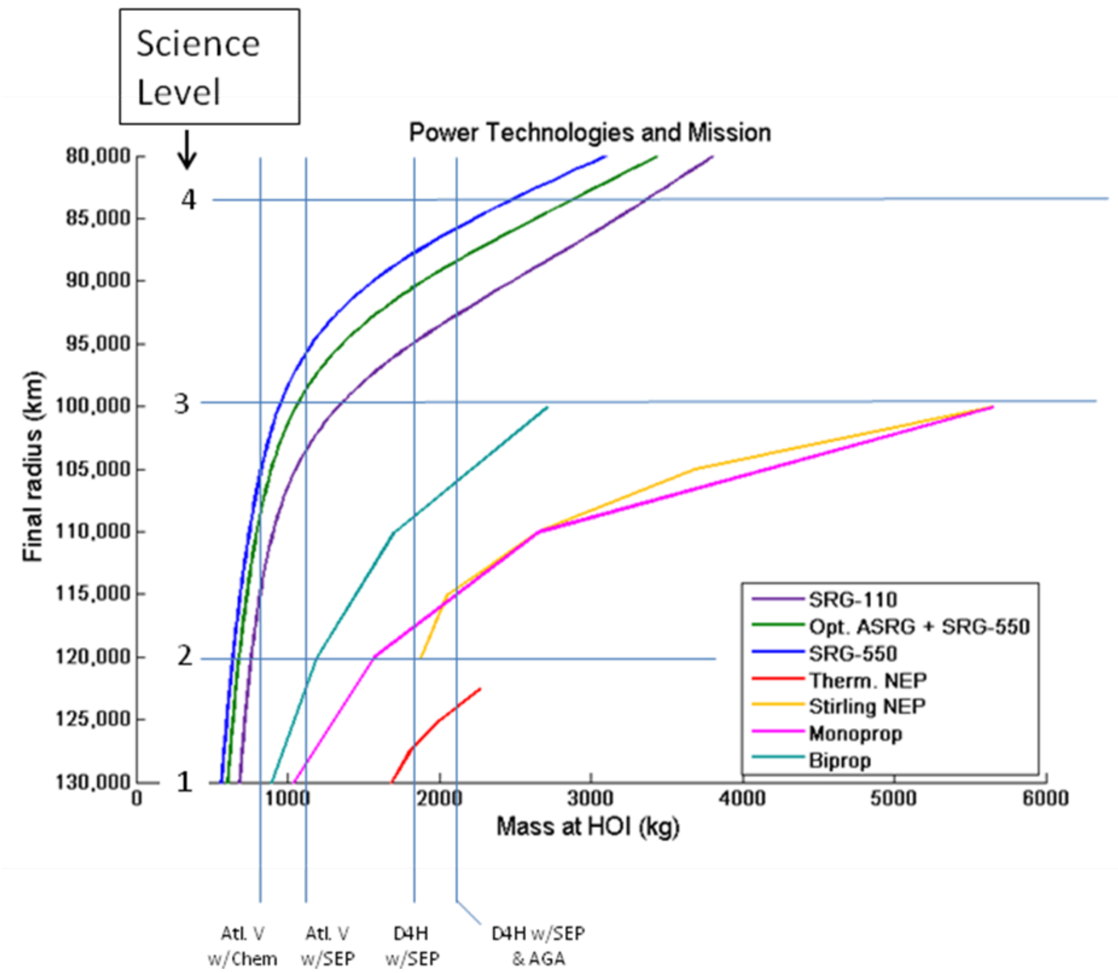

Figure 5-1. Performance curves for various power sources. Launch vehicle performance does not include potential enhancement by Titan AGA or high-I_{SP} chemical propulsion.

Attaining level 4 science also requires propulsion technologies outside of those applicable directly to the hover orbit to deliver sufficient mass to HOI for a level 4 mission. There are two approaches to increasing mass delivered to HOI, applicable to two different mission phases: increase the mass delivered to Saturn approach (launch/transfer phase) or increase the fraction of the Saturn approach mass delivered to HOI (SOI/pumpdown phase). These are not mutually exclusive, so both could be employed for a very demanding mission. A level 4 mission would indeed need both. An effective means of increasing the mass delivered to Saturn approach, and also greatly expanding the suite of potential launch windows beyond years when JGAs are available, is SEP. At power levels larger than those used to date for planetary missions, a 10–15 kW SEP stage (or kit) could add significantly to the delivered mass capability of either Atlas or Delta IV launch vehicles, and would have cross-cutting application potential for many other missions. Technologies that would increase the mass fraction delivered to HOI from Saturn approach address the mission's delta-V-intensive SOI/pumpdown phase. Two technologies that appear effective in this regard are high-I_{SP} in-space chemical propulsion systems, and AGA at Titan. The high-I_{SP} propulsion approach reduces the fraction of the Saturn approach mass dedicated to the SOI/pumpdown stage by significantly decreasing the mass of propellant needed for the required delta-V. One possible implementation would be space-storable cryogenic propellants, dismissed for many years as impractical but recently discussed in the literature [20]. The AGA approach would use a pass through Titan's atmosphere, protected and steered by an aeroshell, to provide delta-V that would otherwise come from propellants. The mass of the aeroshell and other system components would be significantly less than the mass of propellant needed to provide the same delta-V, so the mass savings could be passed on to the hover spacecraft.

Jupiter gravity assist flybys around 2041 could be useful to reduce trip time or increase delivered mass for a prospective SRO mission. Launches to take advantage of that alignment window could occur from 2032 to 2038, with the earlier ones generally associated with larger delivered masses. Full exploration of JGA trajectory space, using inner solar system gravity assists to reach Jupiter, is a difficult task with many subtle nuances, so this study's trajectory search cannot be considered comprehensive. Further exploration of the trajectory space would be worthwhile.

Estimated mission data volumes vary depending on the frequency of image acquisition, but in all cases outstrip the rates available using one 8-hour pass per day with 35 W RF at Ka-band, through a 2.5-m HGA to a single 34-m DSN station. Moving to 50 or even 100 W RF, through a 3 or 4-m HGA, to multiple arrayed 34-m stations greatly increases the available data rates. As something of an upper limit, 100 W RF through a 4-m HGA to four arrayed 34-m ground stations would yield ~1.1 Mbps. This is still far short of meeting the most demanding data acquisition scenarios; therefore, telecommunication advances would be useful.

A surprising result from the study is that at the lower science levels, the capability to fly this mission is closer to achievable than previously thought. At science levels 1 and 2, there are mission concepts that could fly with almost-ready power technology, incremental improvements propulsion technology, and the operations technology. Bringing the ASRG to flight readiness fulfills the power system need. Increasing the global efficiency (jet power divided by power out of the power source, so including PPU efficiency) of low-power electric thrusters, like small Hall thrusters, to 40% or more, fulfills the propulsion need. Operations technologies, especially the ranging and navigation technologies, appear to be the ones needing the most effort for these low science level missions.

Appendix A. Acronyms

AGA	aerogravity assist
ASRG	advanced Stirling radioisotope generator
BOL	beginning of life
CBE	current best estimate
CCD	charge coupled device
CDA	Cassini Dust Analyzer
CML	concept maturity level
DoD	Department of Defense
DOE	Department of Energy
EOL	end of life
EOM	end of mission
EP	electric propulsion
ESA	European Space Agency
FOV	field of view
FPS	fission power system
FY	fiscal year
GPHS	general purpose heat source
GRC	Glenn Research Center
HiVHAc	high-voltage Hall accelerator
HOI	hover orbit initiation
IFOV	instantaneous field of view
IR	infrared
JPL	Jet Propulsion Laboratory
L/D	lift/drag
LIDAR	light detection and ranging
MEL	master equipment list
MEV	maximum expected value
MSL	Mars Science Laboratory
MMRTG	multimission radioisotope thermoelectric generator
NEP	nuclear electric propulsion
NEXT	NASA Evolutionary Xenon Thruster
NRC	National Research Council
POC	point of contact
PPU	power processing unit
PRM	periapse raise maneuver
REP	radioisotope electric propulsion
ROM	rough-order-of-magnitude
RPS	radioisotope power systems
RPWS	Radio/Plasma Wave Science
R_s	one Saturn radius
RTG	radioisotope thermoelectric generator
SAM	Sample Analysis Module
SEP	solar electric propulsion
SMD	Science Mission Directorate
SOA	state of art
SOI	Saturn orbit insertion
SRG	Stirling radioisotope generator
SRO	Saturn Ring Observer
TEGA	Thermally Evolved Gas Analyser
TOF	time of flight
T/P	thrust-to-power
TRL	technology readiness level
TSSM	Titan Saturn System Mission
UV	ultraviolet

Appendix B. References and Bibliography

References

[1] *GPP SRO Questionnaire: Request for Technology Study: Giant Planets Panel, Saturn Ring Observer.* Philip Nicholson, Cornell University, April 2010.

[2] National Aeronautics and Space Administration, and European Space Agency. *Titan Saturn System Mission Final Report on the NASA Contribution to a Joint Mission with ESA.* Task Order #NM0710851, January 30, 2009

[3] Biltgen, P., and T. Ender. "Development of a Collaborative Capability-Based Tradeoff Environment for Complex System Architectures," 44th AIAA Aerospace Sciences Meeting and Exhibit, January 9–12, 2006, Reno, NV, AIAA 2006-728.

[4] Marchak, F., and D. Zulager. "The Effectiveness of Dynamic Graphics in Revealing Structure in Multivariate Data," *Behavior, Research Methods, Instruments and Computers*, 24(2): 253–257, 1992.

[5] Stump, G. M., M. Yukish, T. W. Simpson, and J. J. O'Hara. "Trade space exploration of satellite datasets using a design by shopping paradigm," IEEE Aerospace Conference, 2004.

[6] Lumme, Kari, "On Photometric Properties of Saturn's Rings," *Astrophysics and Space Science* 8: 90–101, 1970.

[7] McGhee, C., et al. "HST Observations of spokes in Saturn's B ring," *Icarus* 173: 508–521, 2005.

[8] Spilker, T. R., "Saturn Ring Observer," 4th IAA Conference on Low-Cost Planetary Missions, IAA-L-0604, May 2000.

[9] Lockheed Martin Energy Systems, ASRG Functional Description, 912/C002144, Rev. 5, December 2009.

[10] National Aeronautics and Space Administration, *Ground Rules for Mission Concept Studies in Support of Planetary Decadal Survey, Revision 2*, Released November 10, 2009.

[11] Schmitz, P., B. Penswick, and R. Shaltens. "Stirling Isotope Power Systems for Stationary and Mobile Lunar Applications," Fourth International Energy Conversion Engineering Conference, AIAA–2006–4036, June 26–29, 2006.

[12] U. S. Department of Energy, Report to Congress, "Startup Plan for Plutonium-238 Production for Radioisotope Power Systems," June 2010.

[13] National Aeronautics and Space Administration, and Department of Energy. *Small Fission Power System Feasibility Study Final Report*, May 13, 2010.

[14] Spilker, T. R. "Saturn Ring Observer," *Acta Astronautica*, 52: 259–265, 2003.

[15] Lockwood, M. K., K. T. Edquist, B. R. Starr, B. R. Hollis, G. A. Hrinda, R. W. Bailey, J. L. Hall, T. R. Spilker, M. A. Noca, N. O'Kongo, et al. "Aerocapture Systems Analysis for a Neptune Mission," NASA/TM-2006-214300, April 2006; published in *Scientific and Technical Aerospace Reports: A Biweekly Publication of the National Aeronautics and Space Administration*, 44(9), May 5, 2006.

[16] The Johns Hopkins University Applied Physics Laboratory, *Titan Explorer Flagship Mission Phase 1 Study Report —2007*, January 2008.

[17] Strange, N. J., and J. M. Longuski. "Graphical Method for Gravity-Assist Trajectory Design." *Journal of Spacecraft and Rockets*, 39(1), 2002.

[18] Casoliva, J. and Lyons, D. T. "Mission Design of Guided Aero-Gravity Assist Trajectories at Titan" Paper AAS 09-432, AAS/AIAA Astrodynamics, Specialist Conference, Pittsburgh, Pennsylvania, August 9–13, 2009.

[19] Campagnola, S., N. J. Strange, and R. P. Russell. "A Fast Tour Design Method Using Non-Tangent V-Infinity Leveraging Transfers," Paper AAS 10-164, AAS/AIAA Space Flight Mechanics Meeting, San Diego, CA, February 2010.

[20] Guernsey, C. S., R. S. Baker, D. Plachta, and P. Kittel, "Cryogenic Propulsion With Zero Boil-off Storage Applied to Outer Planetary Exploration," AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Tuscon, AZ, July 10–13, 2005; available at http://hdl.handle.net/2014/37794.

Bibliography

Nock, K.T. "Rendezvous with Saturn's Rings," *Planetary Rings/Anneaux des Planetes: IAU Colloquium No. 75*, edited by A. Brahic, Toulouse: CNES/IAU (ISBN 2.85428.100.4), pp. 743–759, 1984.

Appendix C. Ring Hazards

The following hazard regions have been identified by the science team. The Saturn Ring Observer (SRO) study has taken these hazards into consideration in the development of the SRO mission concept (e.g., “skipping” over the F ring).

- F Ring : Skip over F ring, ±15 km vertical amplitude and ~10 km vertical thickness
- Pan and Daphnis: Radii of 15 km and 4 km
- Keeler gap edge in vicinity of Daphnis (2–4 km)
- B ring outer edge (117,500 km) vertically scalloped, ~3.5 km
- Inclined and/or puffy ringlets
    - Encke gap ringlets (133,590 km), up to 10 km, small optical depth
    - Charming ringlet (119,940 km), inclined by 3 km
    - Herschel ringlet (118,243 km), inclined few km?
    - “Strange" ringlet in the Huygens Gap (117,735 km) inclined by a few km?

Figure C-1 identifies these hazards based on their radius and relative optical depth.

Furthermore, additional hazards have been identified by the science team that should be noted for a more complete description of the hazards in the ring environment. These include:

- Bending Waves (warping of ring plane)
    - Largest bending wave: Mimas 5:3 vertical resonance, amplitude of ~800 meters
- Splashing within Spiral density waves:
    - Outer A Ring, vertical extent probably only a few tens of meters?
- Impact clouds: could happen at any time
    - Size of particles unknown, vertical extent impact-dependent
- Spokes in B Ring
    - Sizes up to a few microns, may not be a hazard
    - Vertical extent unknown

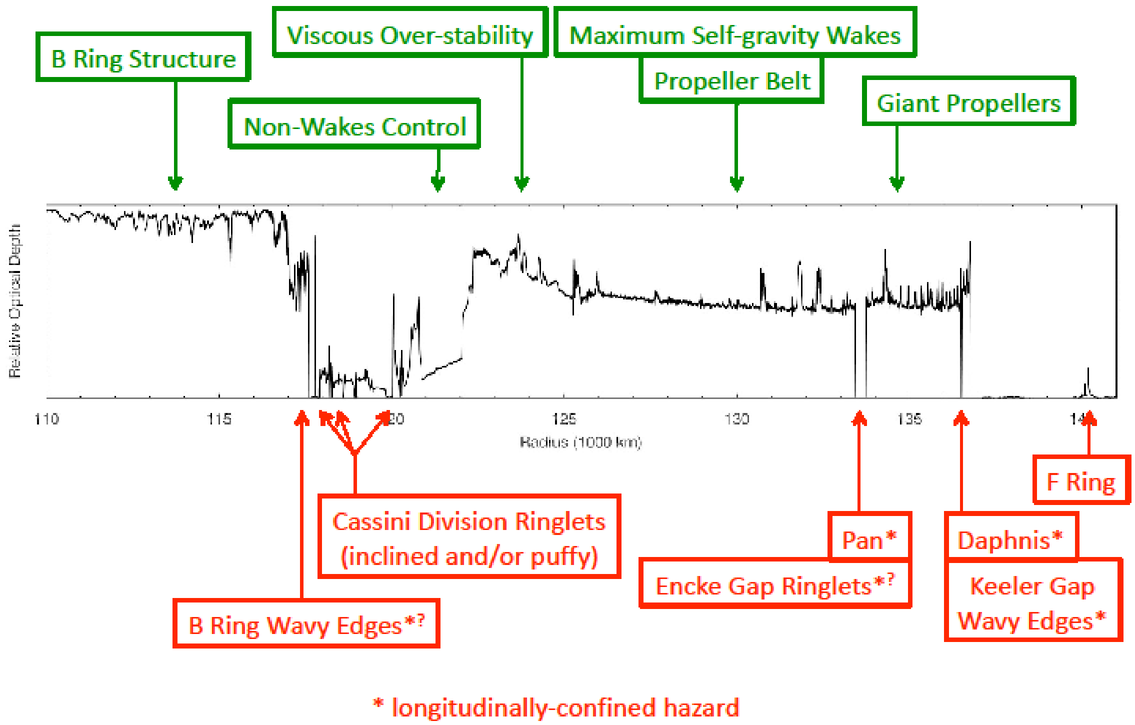

Figure C-1. Targets and Hazards in Saturn's Rings